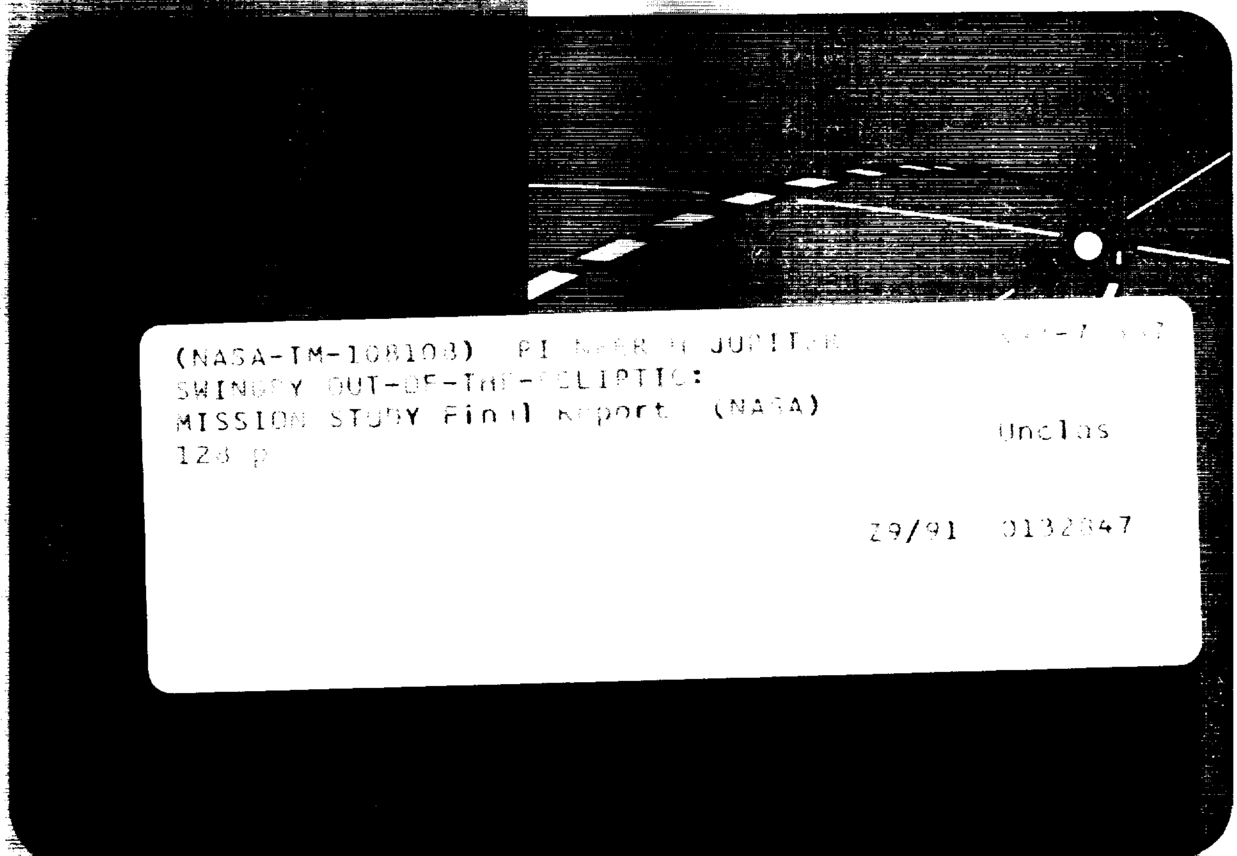
NASA-TM-108108

PIONEER H
JUPITER SWINGBY
OUT-OF-THE-ECLIPTIC

MISSION STUDY



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FINAL REPORT

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PIONEER

OUT-OF-ECLIPTIC MISSION STUDY

AUGUST 20, 1971

FINAL REPORT

Approved: B.D. Padrick
B. D. Padrick, Chief, Advanced Space Projects Office

Approved: John V Foster
J. V. Foster, Director of Development

Prepared by:
R. A. Christiansen
R. W. Jackson
J. Sperans
T. Tendeland
Advanced Space Projects Office

Science Objectives and Rationale provided by:
J. H. Wolfe
Space Sciences Division

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Ames Research Center
Moffett Field, California 94035

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INTRODUCTION

A study of the Pioneer Out-of-Ecliptic Mission was accomplished as an in-house ARC effort, with supporting tasks performed by the TRW Systems Group.

Study objectives were set as follows:

a. Develop the basic scientific rationale for the mission.

b. Establish mission requirements, the range of mission possibilities, and basic mission design.

c. Determine the technical suitability of the present Pioneer for this mission, and particularly, if the F/G prototype spacecraft could be used.

d. Investigate program feasibility for a launch during the 1974 opportunity.

This report has been prepared to present the study results, and the technical basis and background supporting the study conclusions.

The initial NASA Headquarters guidelines specified use of the Titan/Centaur launch vehicle (with appropriate upper stage). Recently, some questions have developed concerning the availability of this booster for a 1974 launch. With this, the study was expanded to consider other launch vehicles, particularly the Atlas/Centaur/TE 364-4. Because of this circumstance, mission characteristics for both Atlas and Titan trajectories are presented. Also, the analyses of Pioneer F/G capabilities are given for the Atlas launched missions as well as Titan.

DEFINITIONS AND ABBREVIATIONS

Ecliptic Plane	the plane of Earth's orbit around the Sun
Ascending Node	the node at which the spacecraft moves from the South hemisphere into the North hemisphere
Line of Nodes	the line of intersection of two planes such as the Ecliptic Plane and trajectory plane
Inclination	the angle between two planes, measured at the ascending node
Swingby	the time period when the spacecraft is near Jupiter
Υ	Vernal Equinox of First Point of Aries - the reference direction in the Ecliptic Plane for measuring longitude
Jupiter departure velocity	the net heliocentric velocity of the spacecraft after swingby which is the vector sum of Jupiter heliocentric orbit velocity and the spacecraft hyperbolic departure velocity
Sun-Spacecraft-Earth Angle	the angle between the Earth-Spacecraft line and the Sun-Spacecraft line
DLA	declination of the launch asymptote and also the greatest inclination required for the near-earth departure trajectory orbit plane
RCA	radius of closest approach to Jupiter
ZAP	angle at Jupiter between the hyperbolic approach velocity and the Jupiter-Sun line
Target Plane (R-T plane)	the plane used to describe aiming points at Jupiter. The target plane is perpendicular to the hyperbolic approach velocity

C_3	twice the launch kinetic energy per unit mass with respect to the Earth
JD	Julian Day number
Perihelion	minimum distance from the Sun
Hyperbolic Approach Velocity	the square root of the approach energy per unit mass with respect to Jupiter. Also the approach velocity with respect to Jupiter at the boundary of the Jupiter sphere of influence
Jupiter Sphere of Influence	the region of space around Jupiter where the spacecraft trajectory is primarily influenced by Jupiter's gravity. The Jupiter sphere of influence extends to about 675 Jupiter radii from the planet.
Hyperbolic Departure Velocity	the departure velocity of the spacecraft with respect to Jupiter at the boundary of the Jupiter sphere of influence

1.0 SUMMARY AND CONCLUSIONS

1.1 PRINCIPAL SCIENTIFIC OBJECTIVES

a. Investigation of the large scale flow pattern of the solar wind and the coronal expansion process.

b. Investigation of the interplanetary magnetic field with emphasis on the question of field sector structure existence at high solar latitudes.

c. Investigation of solar cosmic rays with regard to production, storage, and propagation at high solar latitudes.

d. Measurements of Galactic cosmic ray intensity, with the possibility of observing conditions over the solar poles, which closely resemble interstellar intensities.

e. Additional planetary observations during Jupiter swingby.

1.2 MISSION UNIQUE FEATURES

a. The first opportunity for in-situ, three dimensional exploration of the Heliosphere, heretofore limited to the region of the ecliptic plane.

b. The first direct observations of the Sun at high solar latitudes, with the opportunity to compare studies of interplanetary phenomena and those associated solar phenomena at the higher latitudes.

c. The first opportunity for a nonequatorial solar radio occultation.

1.3 MISSION ANALYSIS

1.3.1 Titan Launched Mission

a. The Titan/Centaur/TE 364-4 vehicle can launch a Pioneer spacecraft with sufficient energy to achieve high ex-ecliptic inclinations with trajectory passage directly over the solar poles.

b. Jupiter targeting for a Titan launched mission, to achieve an optimum post-encounter Out-of-Ecliptic trajectory results in a 3.2 R_J swingby distance (from planet center). Targeting at greater distances from Jupiter will still provide high inclinations, but with increased trip time.

c. Time required for the nominal mission from launch to passage around and over the second solar pole (North Sun pole first) is approximately 3.7 years.

1.3.2 Atlas Launched Mission

a. The Atlas/Centaur/TE 364-4 vehicle with a Pioneer payload can achieve an inclination of approximately 42 degrees with respect to the ecliptic plane. Maximum solar latitudes reached would be near 45 degrees.

b. Jupiter targeting for the lower arrival velocity of an Atlas trajectory requires a swingby distance at Jupiter of nearly 20 R_j.

c. Time required for the nominal Atlas mission is 4.6 years.

1.3.3 General

a. The 1975 and 1976 launch opportunities compare closely with the 1974 opportunity.

b. The Titan trajectory satisfies all mission and science requirements.

c. The Atlas trajectory provides an opportunity for new interplanetary and solar explorations at large distances Out-of-the-Ecliptic plane, but does not achieve solar pole crossings.

1.4 SPACECRAFT ANALYSIS

a. It is feasible and cost effective to utilize the Pioneer F/G prototype spacecraft for an Out-of-Ecliptic launch in 1974.

b. Spacecraft capabilities are compatible with both Titan and Atlas mission requirements. Scientific mission objectives can be accomplished without change to the present F/G spacecraft design.

c. The spacecraft has been qualified for both Atlas/Centaur/ TE 364-4 and Titan/Centaur/TE 364-4 launch environments.

d. RTG performance determines ultimate spacecraft operating lifetime. A five year mission duration can be achieved with nominal RTG performance.

e. The Out-of-Ecliptic spacecraft will experience a lesser radiation exposure at Jupiter than is expected for the Pioneer F encounter, however, the F measurements will be important in determining final Out-of-Ecliptic planetary swingby distance.

f. Design modifications have been investigated for extended mission life and increased mission reliability.

1.5 SCIENCE INSTRUMENT REQUIREMENTS AND PAYLOAD OPTIONS

a. The Pioneer F/G instrument payload can satisfy scientific mission objectives.

b. A full set of spare instruments are being produced for the F/G program which could be used as the flight instruments for an Out-of-Ecliptic mission.

c. A number of payload options exist, ranging from internal modifications of present instruments to the addition of new instruments, which are considered to be feasible for a 1974 launch.

1.6 PROGRAM IMPLEMENTATION SCHEDULES

a. A program start date in FY 1972 is required for launch in 1974.

b. A 1974 launch date is based on using the F/G prototype spacecraft and the F/G instrument spares. The schedule would allow some limited changes to the spacecraft and/or the science payload.

c. No increase in ARC manpower or resources is required for the program. Activity would phase compatibly with the F and G effort.

d. A system contract with TRW is assumed.

2.0 SCIENTIFIC RATIONALE AND OBJECTIVES

The primary objective of the Pioneer Out-of-Ecliptic mission is to make the first exploratory investigation of the interplanetary medium away from the ecliptic plane. All measurements of the interplanetary medium up to now, have been made within a few degrees of the ecliptic plane, and have provided only a two dimensional picture.

The mission will provide a unique opportunity for studies of interplanetary and solar phenomena at high heliographic latitudes.

The mission will provide the first opportunity to examine galactic and solar cosmic rays at high heliographic latitudes.

The mission presents an opportunity to perform a non-equatorial radio occultation of the spacecraft by the Sun.

The mission will also afford additional scientific observations of Jupiter during the swingby maneuver.

2.1 LARGE SCALE STRUCTURE

An important objective for the Out-of-the-Ecliptic mission is the determination of the large scale flow pattern of the solar wind. The solar wind is the dominant driving force from the Sun which influences the planets, aside from the electromagnetic radiation. The connection between the solar wind and the Sun has only been made by spacecraft orbiting between Venus and Mars, and only in the plane of the ecliptic. Since the surface disk of the Sun reveals many latitude dependent features, it is most likely that these will be reflected in the solar wind at high heliographic latitudes. In the spherical expansion model of the solar wind, the gas flows radially outward from the Sun equally in all directions and cools adiabatically as it expands. The Out-of-Ecliptic mission will provide the opportunity for determining not only the validity of this model but the ramifications of the large scale structure to the fine features in the particles and fields environment and to solar-interplanetary relationships. In addition, the understanding of the total coronal expansion process requires the measurement of the latitude dependence of the flow parameters such as velocity, density and temperature. An important question related to the large scale dynamics of the interplanetary medium concerns the degree to which the solar wind stream/stream interactions are inhibited at high solar latitudes. The Out-of-the-Ecliptic mission also affords the opportunity for comparative studies of the inner coronal structure via a non-equatorial solar radio occultation measurement.

2.2 MAGNETIC FIELDS

At present, there is conflicting evidence with regard to the degree to which the interplanetary magnetic field sector structure persists to high solar latitudes. Related to the solar wind flow pattern is the question of whether the field has a non-zero polar component. The determination of the latitude dependence of waves, discontinuities and disturbances is important in order to assess the solar-interplanetary relationships at high latitudes. Finally, there is the intriguing question as to whether magnetic field lines emitted near the poles of the Sun connect directly to the interstellar medium.

Measurements obtained from the OGO 5 and Mariner spacecraft by Rosenberg and Coleman (1969) indicate a 75% dominance of one polarity at 7.3 degrees heliographic latitude. Specifically, they find that the dominant polarity of the interplanetary field was inward at heliographic latitudes above the solar equatorial plane and outward at latitudes below this plane. These measurements are consistent with a dipolar component of the solar magnetic field. These findings are, however, in direct conflict with high latitude magnetograph measurements which show a positive correlation with the measured interplanetary magnetic field sector structure.

Measurements obtained by Coleman and Rosenberg (1971) and Rosenberg, Coleman and Colburn (in press) indicate the presence of a non-zero polar component in the interplanetary magnetic field generally on the order of 0.3γ at ± 7.5 degrees heliographic latitude. In the solar equatorial plane they find this component goes to zero. The measurements suggest that the interplanetary field lines expand away from the solar equatorial plane. Latitude effects in the solar wind have been confirmed by the Vela spacecraft (Hundhausen, Asilomar Conference, in press) which generally show a lower density and higher velocity off the equatorial plane as compared to in this plane.

2.3 PRODUCTION, STORAGE AND PROPAGATION OF SOLAR COSMIC RAYS

At the present time questions relating to the heliographic latitude dependence of the production, storage and propagation of solar cosmic rays are completely open. For example, are solar cosmic rays even produced at high latitudes? In addition, can flare particles produced at lower latitudes diffuse poleward and be emitted from the Sun on high latitude field lines. Measure of the distribution of solar cosmic rays in the polar direction during flare events will be useful for determining how solar magnetic fields near active regions connect onto fields leading to the interplanetary medium at high latitudes.

2.4 GALACTIC COSMIC RAYS

To gain access to the inner solar system over the poles, galactic cosmic rays must propagate a relatively short distance along magnetic lines of force which are barely spiralled. On the other hand, in the Ecliptic, particles must propagate a considerable distance along fields that are executing a tight Archimedes spiral pattern. Solar activity is noticeably less intense near the poles than it is at lower latitudes and the stream/stream interactions should tend to be less at the higher latitudes such that the galactic cosmic ray modulation would be considerably less over the solar poles and be neither isotropic nor homogeneous. It is conceivable that galactic cosmic ray intensities that one could observe over the poles may resemble quite closely the interstellar intensities, while those in the ecliptic will be heavily modulated even out to considerable radial distances. Observations of the distribution of galactic cosmic rays during Forbush Decreases will be helpful in determining the extent to which interplanetary shock disturbances extend in solar latitude.

2.5 SOLAR CYCLE ACTIVITY RELATED TO MISSION PHASES

Figure 2-1 shows the Maunder Butterfly Diagram of the heliographic position of sunspots as a function of time through the solar cycle. The proposed 1974 launch date, Jupiter encounter and passage over the northern and southern solar poles are indicated on the diagram. As is seen in the beginning of the cycle the spots appear at approximately ±30 to 35 degrees in solar latitude and as the cycle progresses, the appearance of the spot groups continuously moves closer and closer to the equator and is within a few degrees of the equator at the end of the cycle. The observation of the latitude dependence of the spot phenomena thus requires a heliographic latitude excursion of at least ±35 degrees.

2.6 SCIENCE REQUIREMENTS ON THE OUT-OF-ECLIPTIC MISSION DESIGN

There are three principal requirements which the science places on the Out-of-the-Ecliptic mission. The first requirement is a constraint on the trajectory that during the polar passage the spacecraft be located between one to two AU from the Sun. This distance is desirable for comparison studies with exploration of the interplanetary medium between 1-2 AU in the ecliptic which has already been accomplished. Secondly, it is desirable that there be at least six solar rotations during a Pole-to-Pole Passage. This is desirable since a trajectory which carries a spacecraft from +90 degrees to -90 degrees in six solar rotations would smear no more than 30 degrees per solar rotation. A latitude smear of greater than 30 degrees would tend to mask some of the finer details in the latitude dependencies. Finally, the spacecraft orientation must be compatible with the instrument viewing requirements. Of primary concern here is the view angle requirements of the plasma probe that it lie within ±40° of the solar direction.

MAUNDER BUTTERFLY DIAGRAM OF SUNSPOT HELIOGRAPHIC POSITION
[ADAPTED FROM J. C. BRANDT, INTRO. TO THE SOLAR WIND
(INTERSCIENCE, 1970)]

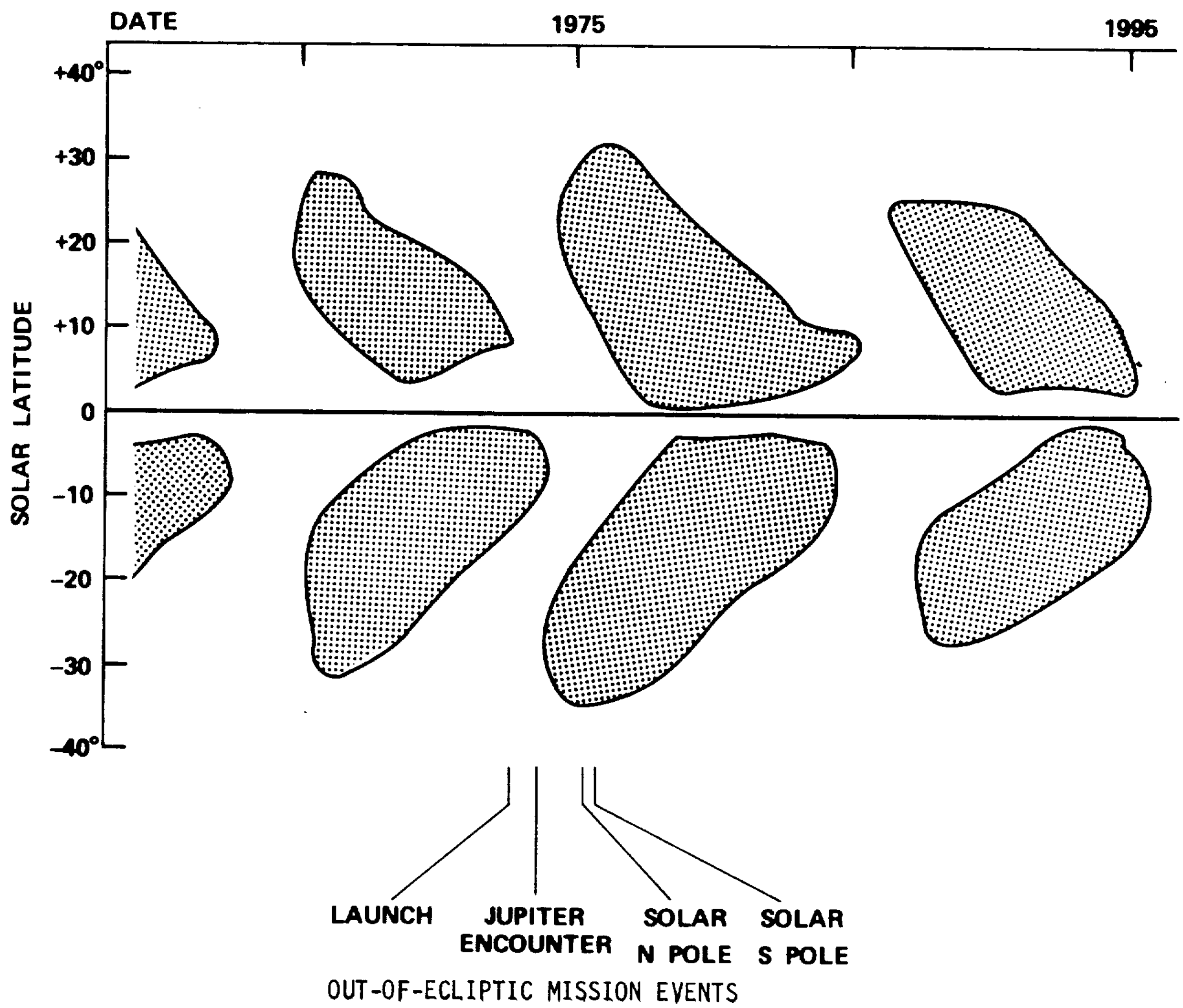

Figure 2-1.- Maunder Butterfly Diagram

Heliographic Position of Sun Spots vs. Time

2.7 PIONEER F/G SCIENCE PAYLOAD SUITABILITY

Table II-1 lists the science instruments presently being carried by the Pioneer F/G spacecraft. The suitability of the various experiments during the transit to Jupiter, the Jupiter encounter itself and the post encounter out of the ecliptic phase is indicated. As can be seen, the particles and fields experiments presently being carried are quite suitable to the mission whereas the planetary experiments may provide some information on the Out-of-the-Ecliptic mission but are primarily designed for the encounter portion. Not shown are the celestial mechanics and radio occultation experiments which utilize the spacecraft S-Band Communications System. The occultation experiment is considered vital in the Out-of-the-Ecliptic mission.

Table II-1

SUITABILITY OF PIONEER F/G INSTRUMENTS FOR PIONEER H MISSION

	F/G Instrument	Transit to Jupiter	Jupiter Encounter	Post Encounter Out-of-Ecliptic
1.	Helium Vector Magnetometer (JPL)	X	X	X
2.	Plasma Analyzer (ARC)	X	X	X
3.	Charged Particle Instrument (U. of Chi.)	X	X	X
4.	Cosmic Ray Telescope (GSFC)	X	X	X
5.	Geiger Tube Telescope (U. of Iowa)	X	X	X
6.	Trapped Radiation Detector (UCSD)	X	X	X
7.	UV Photometer (USC)	Neutral H Only	X	Neutral H Only
8.	IR Radiometer (CIT)		X	
9.	Asteroid/Meteoroid Detector (GE)	X		(Potential)
10.	Meteoroid Detector (LaRC)	X		(Potential)
11.	Imaging Photopolarimeter (U. of Ariz.)	Zodiacal Light Only	X	Zodiacal Light Only

3.0 MISSION ANALYSIS

Mission analysis for the Out-of-Ecliptic mission is divided into two parts, the Earth to Jupiter trajectory and the post swingby trajectory.

The Earth to Jupiter trajectory will be similar to the Pioneer F and G design trajectory and will not constrain the Out-of-Ecliptic Mission.

The mission analysis first describes post swingby trajectory requirements and spacecraft and science constraints. The logic used to choose nominal and alternate Titan/Centaur missions is described.

The nominal and alternate Titan/Centaur missions for 1974 launch are defined.

The use of Atlas/Centaur and Titan 3C launch vehicles is discussed and a nominal Atlas/Centaur mission for 1974 launch is defined.

Launch opportunities for Out-of-Ecliptic missions in later years are also discussed.

3.1 OUT-OF-ECLIPTIC MISSION REQUIREMENTS

3.1.1 General Mission Description

The Out-of-Ecliptic mission will launch a spacecraft to Jupiter and use the planet's gravity field to rotate the trajectory so the spacecraft swings up out of the plane of the ecliptic and back around the Sun. Figure 3-1 shows the mission schematically. Conceptually, an out-of-ecliptic mission could be launched directly from Earth into a high inclination orbit. The velocity required, however, is much greater than any reasonable launch vehicle can provide. By using a Jupiter swingby to provide the Out-of-Ecliptic trajectory, the launch velocity requirements are low enough to be met by Titan Centaur or Atlas Centaur class launch vehicles.

Mission analysis which follows is simplified by the planar characteristic of the out-of-ecliptic post swingby trajectory. The trajectory plane inclination can be considered separately from the in-plane parameters of perihelion distance and trip time.

3.1.2 Post Swingby Trajectory Inclination

The scientific experiment requirements for this mission dictate a trajectory which will achieve the highest possible solar latitudes preferably one which passes directly over the solar poles. The Sun's poles are on its spin axis, which is tilted 7.25° from a perpendicular to the plane of the ecliptic. The spin axis tilt must be taken into account in determining the trajectory plane inclination required to pass over the poles. The tilt also means that trajectories which pass over the North Pole first have a different inclination to the ecliptic than those which pass over the South Pole first.

Since the Out-of-Ecliptic trajectory is established at Jupiter swingby, the position of Jupiter (date of swingby) controls the inclination of planes which pass over the Sun poles. The relation of the Sun poles to Jupiter and the trajectory plane inclinations required are shown in Figure 3-2.

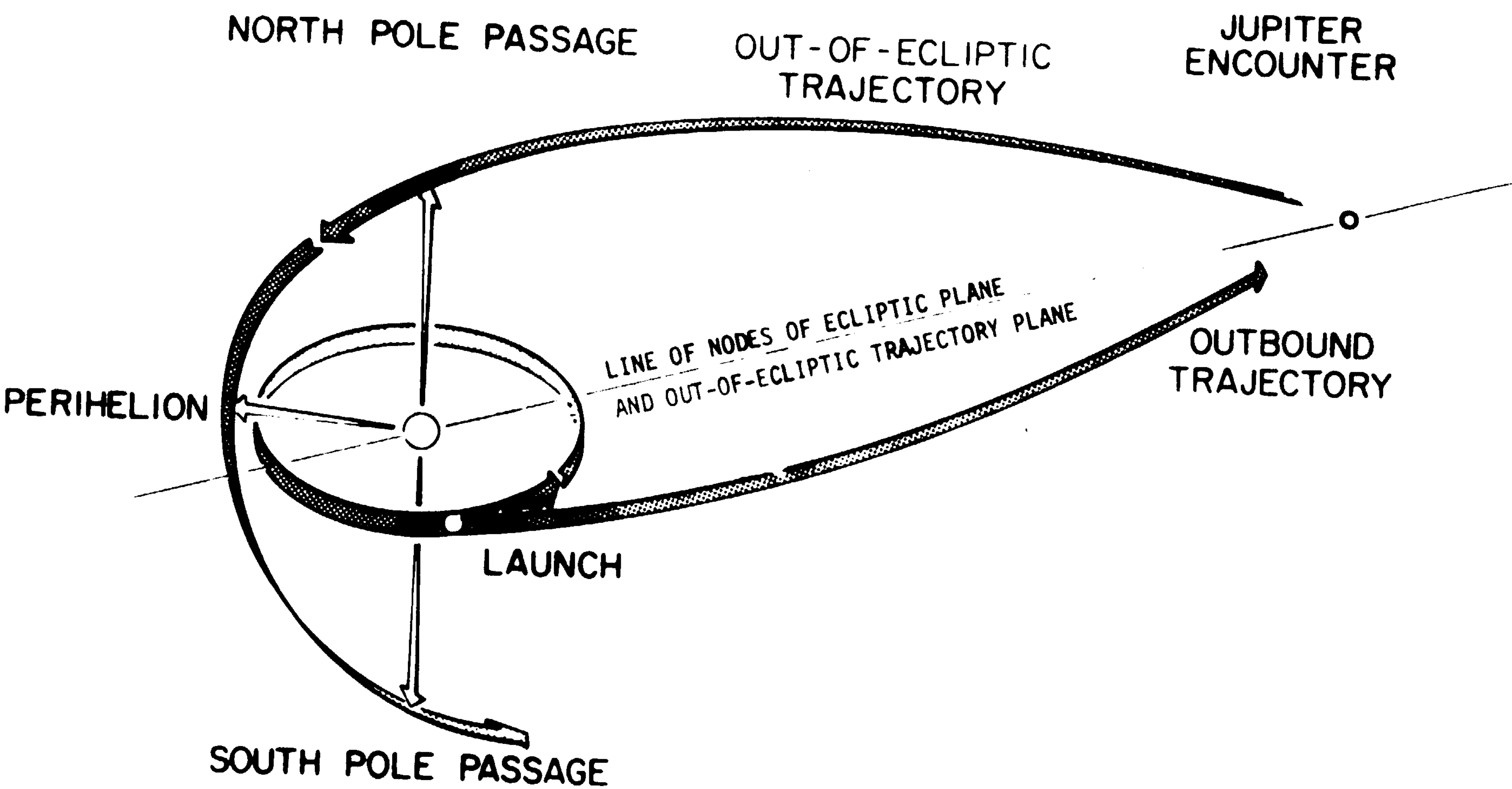

Figure 3-1.- Trajectory for Out-of-Ecliptic Mission

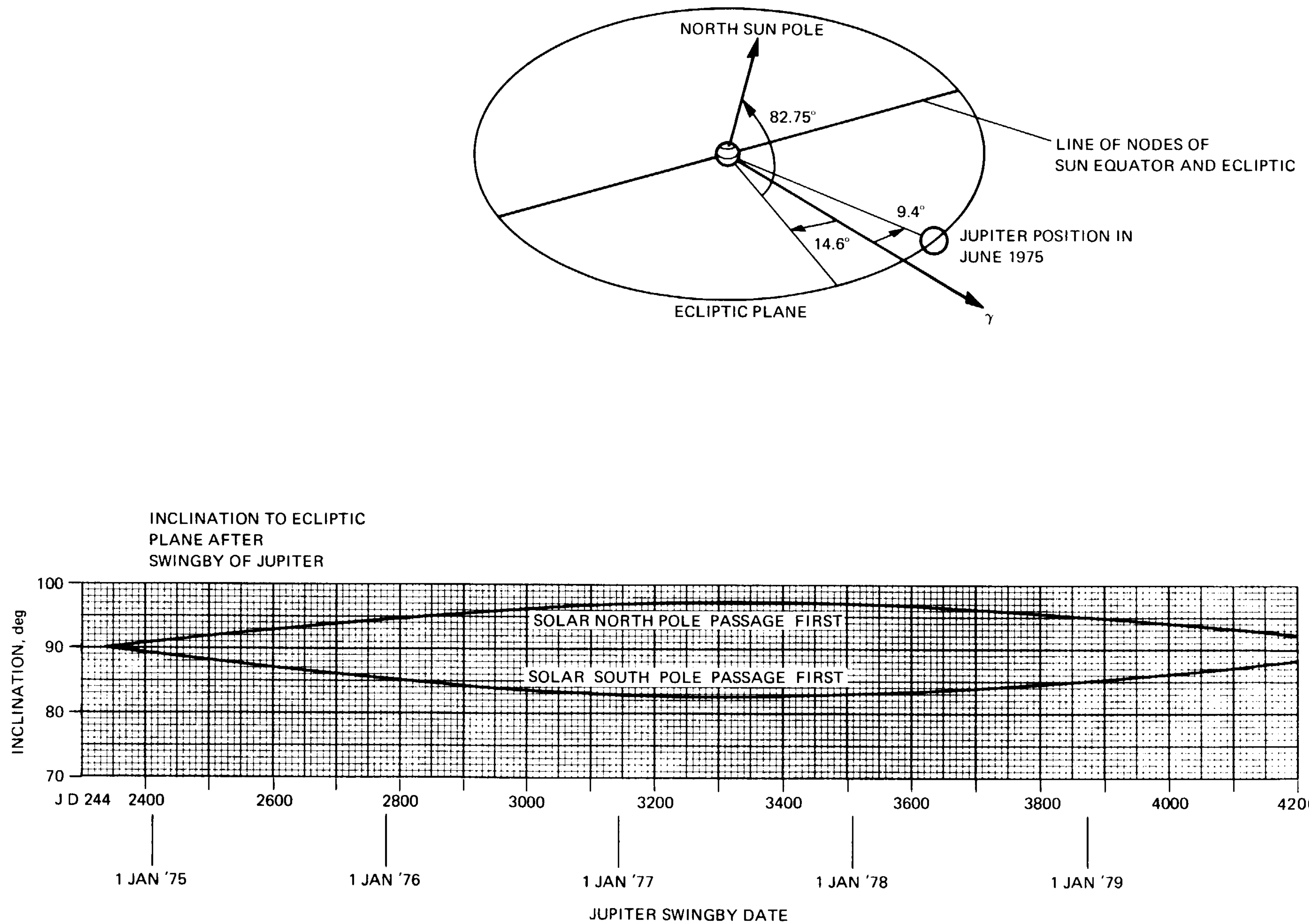

FIGURE 3-2. – TRAJECTORY PLANE INCLINATIONS TO FLY OVER SOLAR POLES

Passage over the North Sun Pole first allows the best mission coverage from the 210' antennas of the DSN. Figure 3-3 shows the tracking duration per day available from the 210' antennas. Notice that a small "hole" of reduced daily coverage exists around 50° S latitude and that a single antenna can provide continuous tracking when the spacecraft latitude is greater than 60°.

For launch during the 1974 opportunity, Jupiter arrival is during June 1975. From Figure 3-2 an inclination of 92.5° is required after swingby to pass directly over the Sun poles, North Pole first.

3.1.3 Jupiter Hyperbolic Approach Velocity Required

The spacecraft is deflected into an Out-of-Ecliptic orbit by the Jupiter gravity field. The spacecraft approaches Jupiter with a particular hyperbolic approach velocity, and after encounter, leaves Jupiter with the same speed but heading in a different direction. Because gravity is a conservative force, the departure velocity has the same magnitude as the arrival velocity, only the direction is changed.

Figure 3-4 includes a sketch showing how the hyperbolic departure velocity is vectorially added to Jupiter's orbit velocity to produce the spacecraft departure velocity. The inclination of the post swingby trajectory plane is the same as the angle between the departure velocity and the ecliptic plane. (Jupiter's orbit velocity is parallel to the ecliptic in June 1975 which simplifies the sketch).

The magnitude and flight path angle of the departure velocity determine the in-plane trajectory parameters such as perihelion distance and trip time.

Jupiter swingby conditions and targeting are used to direct the departure hyperbolic velocity in the desired direction.

Figure 3-4 shows the relation between the post swingby trajectory inclination and the magnitude of the Jupiter hyperbolic approach velocity. The figure is calculated for zero flight path angle (departure velocity perpendicular to the Jupiter-Sun line).

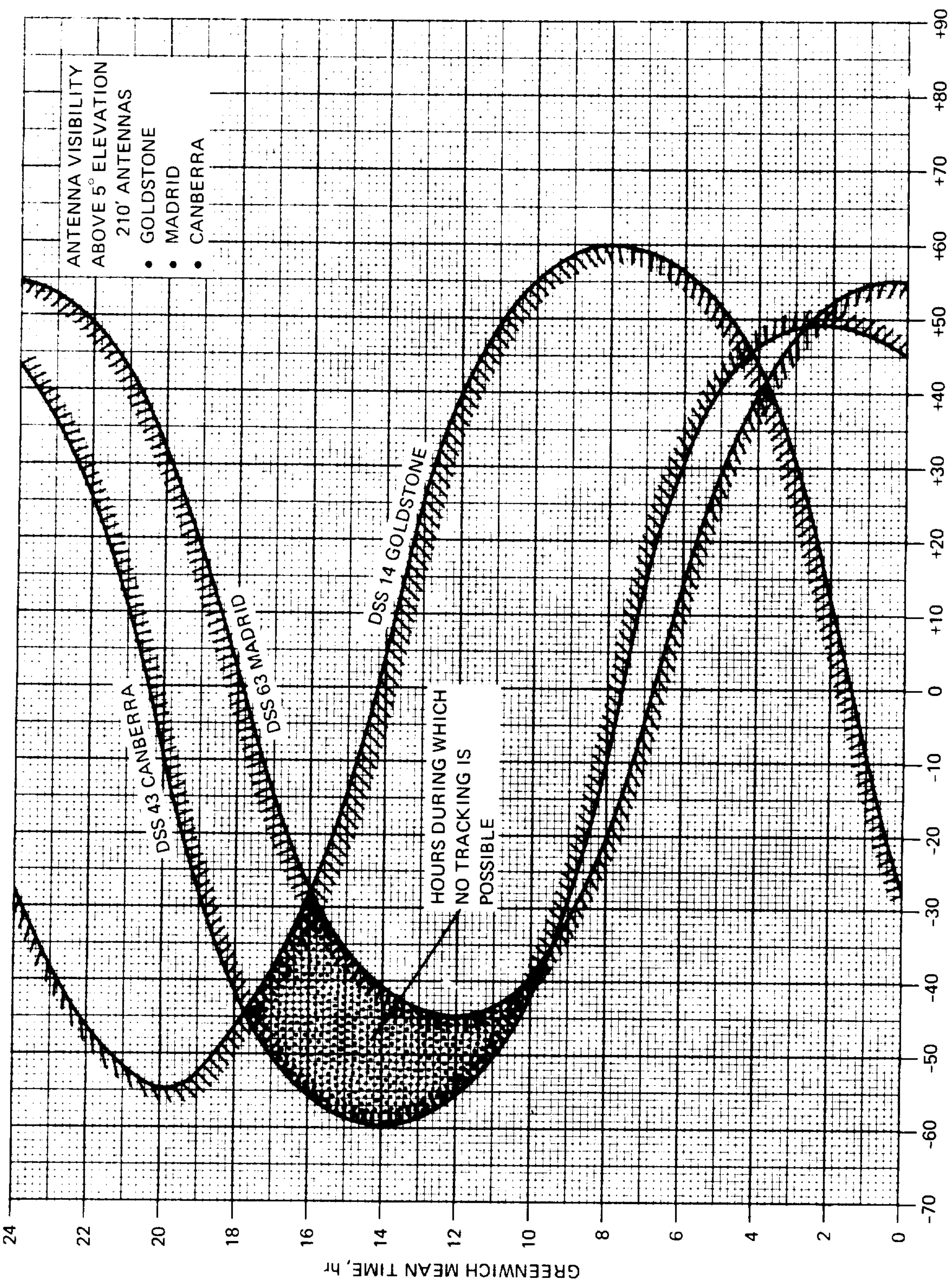

FIGURE 3-3.–DAILY VISIBILITY TIME FROM 210' ANTENNA SITES

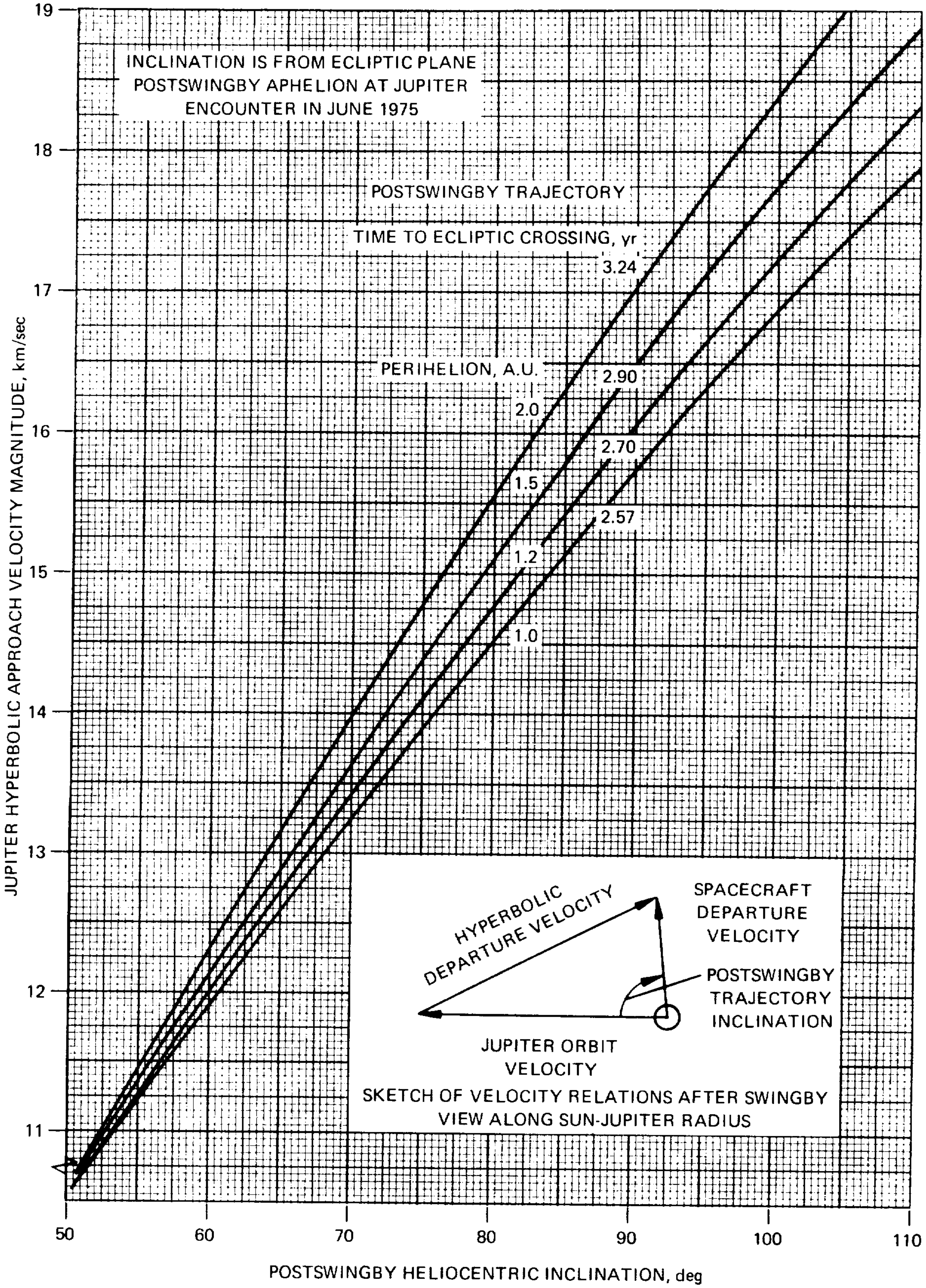

FIGURE 3-4.–POSTSWINGBY HELIOCENTRIC INCLINATION

The Plasma Probe viewing angle for measurements of the Sun sets a maximum of 40° on the Sun-Spacecraft-Earth Angle.

During the trajectory to Jupiter, the Sun-Spacecraft-Earth angle history will be similar to Pioneer F. During the out-of-ecliptic return to the Sun, the spacecraft will be above the plane of the Earth's orbit and the angle history will be different.

The maximum value of the angle will occur after the spacecraft passes over the North pole of the Sun and crosses through the ecliptic plane. Figure 3-6 shows the geometry of the Earth orbit and shows the Earth positions where the maximum Sun-Spacecraft-Earth angle will be less than 40°.

For a 92.5° inclination trajectory, the maximum angle occurs slightly before the spacecraft passes through the ecliptic. The range of Earth positions must be restricted to that shown in figure 3-6.

Since each Earth position corresponds to a time, the permissible Earth positions determine the trip times which may be used from swingby to ecliptic crossing.

For the 396 day trip to Jupiter shown, the arc of good Earth positions corresponds to trip times from swingby to ecliptic crossing of 0.2 to 0.4 years, 1.2 to 1.4 years, 2.2 to 2.4 years, 3.2 to 3.4 years, and 4.2 to 4.4 years.

From figure 3-5 each of these mission durations requires a unique Jupiter hyperbolic approach velocity and swingby RCA.

The 1.2 year trip requires the spacecraft to swingby Jupiter at such a close distance that radiation damage is probable.

The 2.2 year trip, with RCA = 3.2 R_J has been chosen as the Nominal Mission for the Titan/Centaur launch vehicle.

The 3.2 year trip, with RCA = 5.4 R_J has been chosen as the Alternate Mission for the Titan/Centaur launch vehicle.

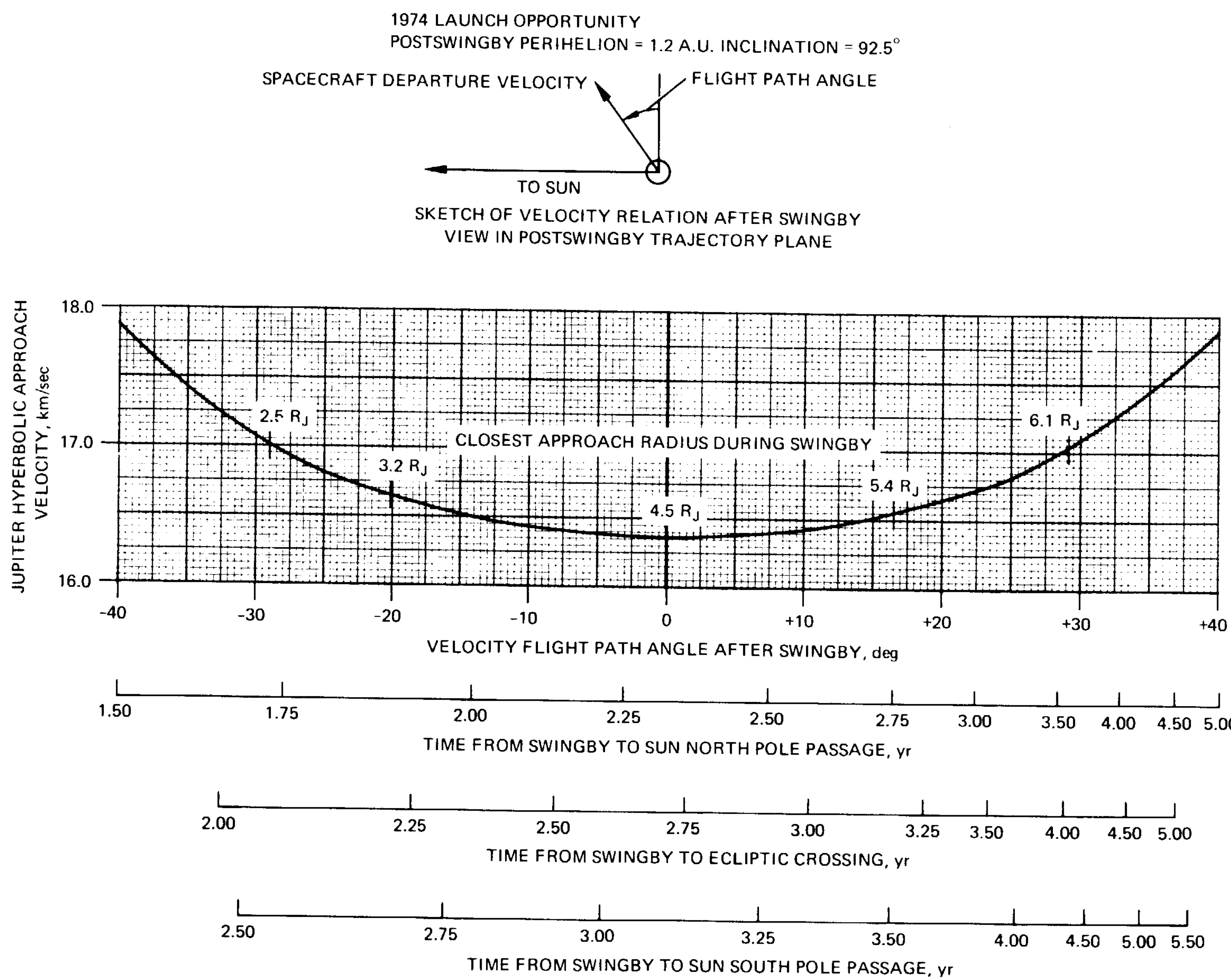

FIGURE 3-5.–TRIP TIMES AFTER JUPITER SWINGBY

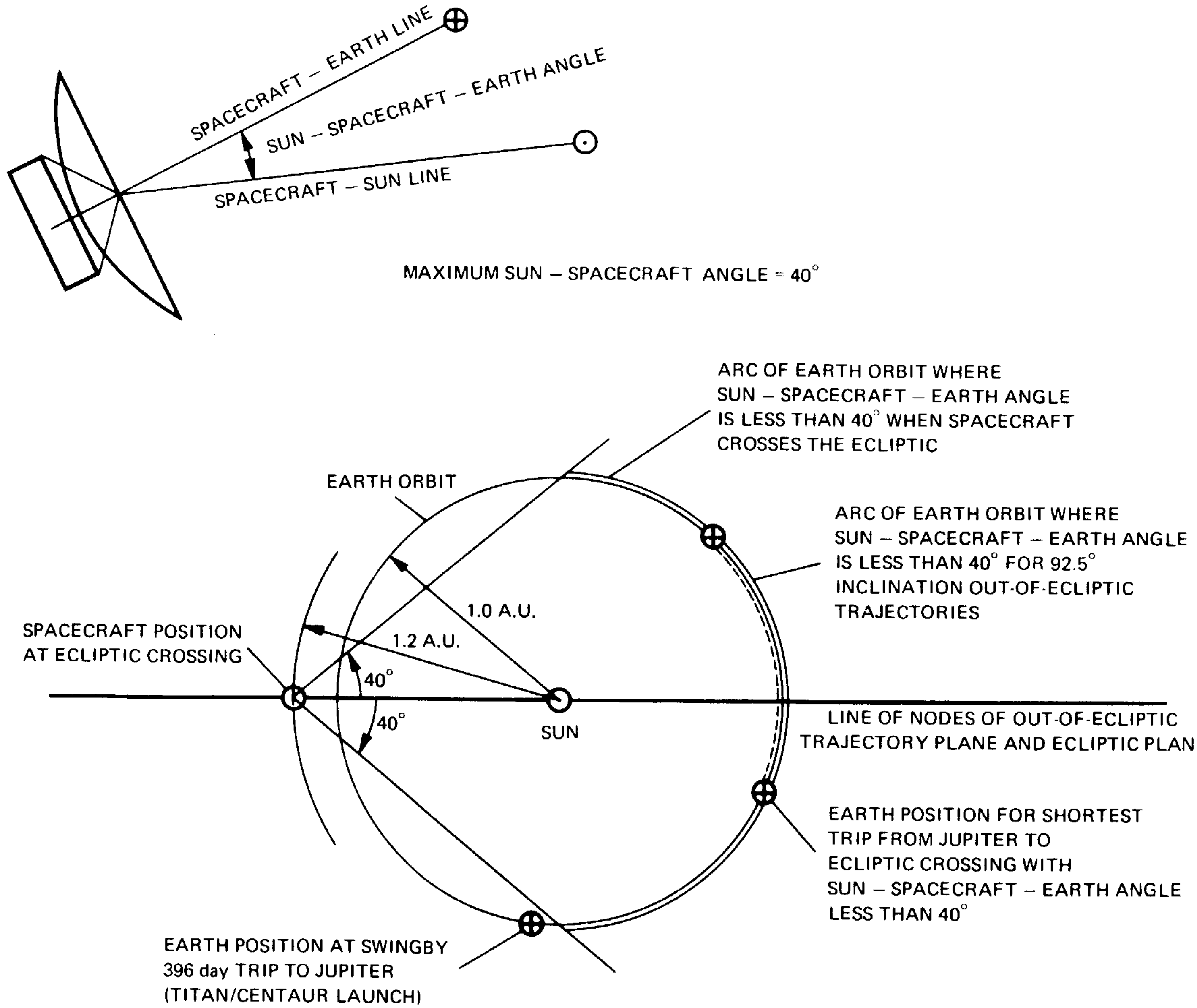

FIGURE 3-6.–SUN – SPACECRAFT – EARTH ANGLE AT ECLIPTIC CROSSING

The figure shows that hyperbolic approach velocities greater than 16.0 km/sec allow 92.5° trajectory inclinations and flexibility in choosing the post swingby perihelion.

3.1.4 Spacecraft Constraints

3.1.4.1 Perihelion Distance

The Pioneer F/G spacecraft is designed to function at 1 AU distance from the Sun immediately after launch. The Out-of-Ecliptic mission will return to the Sun about three years after launch. During that time thermal coatings and paint will degrade and allow more solar energy to be absorbed with higher spacecraft temperatures resulting.

To satisfy scientific requirements for the mission, the perihelion distance should be low, so that spacecraft will pass by the Sun between 1 AU and 2 AU. As a compromise, a perihelion distance of 1.2 AU has been chosen for this study.

Figure 3-4 shows that the hyperbolic approach velocity requirements vary directly with perihelion distance.

3.1.4.2 Trip Time to the Sun

Trip times to the Sun are closely related to the flight path angle of the spacecraft departure velocity after Jupiter swingby. Negative flight path angles allow the spacecraft to pass perihelion after first Sun polar passage and before crossing the ecliptic. This shortens the flight time to the Sun. Positive flight path angles require the spacecraft to pass aphelion before first Sun polar passage and lengthen flight time.

Figure 3-5 shows how flight path angle influences trip times. Note that the approach velocity must be increased as flight path angle changes to maintain the same perihelion distance and inclination. The swingby closest approach points show that the spacecraft must pass closer to Jupiter to achieve short mission times.

3.1.4.3 Sun-Spacecraft-Earth Angle - Trajectory Phasing

The Pioneer F/G spacecraft requires the spin axis to be pointed at Earth for good communication. The Sun-Spacecraft-Earth Angle describes the angle between the spin axis and the Spacecraft-Sun direction.

3.1.4.4 Closest Approach to Jupiter

During the Jupiter encounter, the spacecraft and experiments will pass within Jupiter's radiation belts. Pioneer F is planned to pass Jupiter at a closest approach radius of 3.0 Jupiter radii.

The Nominal Out-of-Ecliptic mission for a Titan/Centaur launch passes by Jupiter at 3.2 Jupiter radii closest approach, and will encounter about the same radiation environment as Pioneer F.

If the radiation belts are measured by Pioneer F to be more intense than expected, the alternate mission may be used which passes by Jupiter at 5.4 Jupiter radii closest approach, and will encounter less radiation than Pioneer F.

Radiation fluences are discussed in more detail in section 4.2.1.3 "Jupiter Radiation Belts".

3.2 1974 MISSION OPPORTUNITY

3.2.1 Launch Window

Since the Out-of-Ecliptic mission includes a Jupiter swingby, it must be launched during a launch window for flights to Jupiter. The launch date-arrival date curve for the 1974 launch opportunity is shown in figure 3-7. Curves are included for Jupiter hyperbolic approach velocity magnitude and trip time to Jupiter.

For Jupiter approach velocities of 16 km/sec, trip times to Jupiter will be about 400 days.

3.2.2 C_3 Requirements

Since the Jupiter hyperbolic approach velocity is important in determining the post swingby trajectory, its relation to Earth launch C_3 is shown in more detail in figure 3-8.

For a 20-day launch window, the C_3 must be at least 147.5 km^2/sec^2 to provide a Jupiter approach velocity of 16 km/sec.

The C_3 requirements for postswingby trajectories which pass over the Sun poles, are close to the value which allows a direct escape from the Solar System. Thus, the trajectories to Jupiter will be hyperbolas or segments of very long ellipses.

3.2.3 Launch Azimuth and Parking Orbit Coast

Figure 3-7 shows a curve of DLA = 34°. The trajectories to Jupiter which will be used are well within the DLA boundary and will not be restricted by permissible launch azimuths.

The launch trajectory will include a coast period in a parking oribt. For launch azimuths between 90° and 110° (Pioneer F limits) the parking orbit coast durations required do not exceed 20 minutes. This is well within the Centaur limit of 30 minutes. Approximately 2 hours are available for launch each day during the launch window for launch aximuths between 90° and 110°.

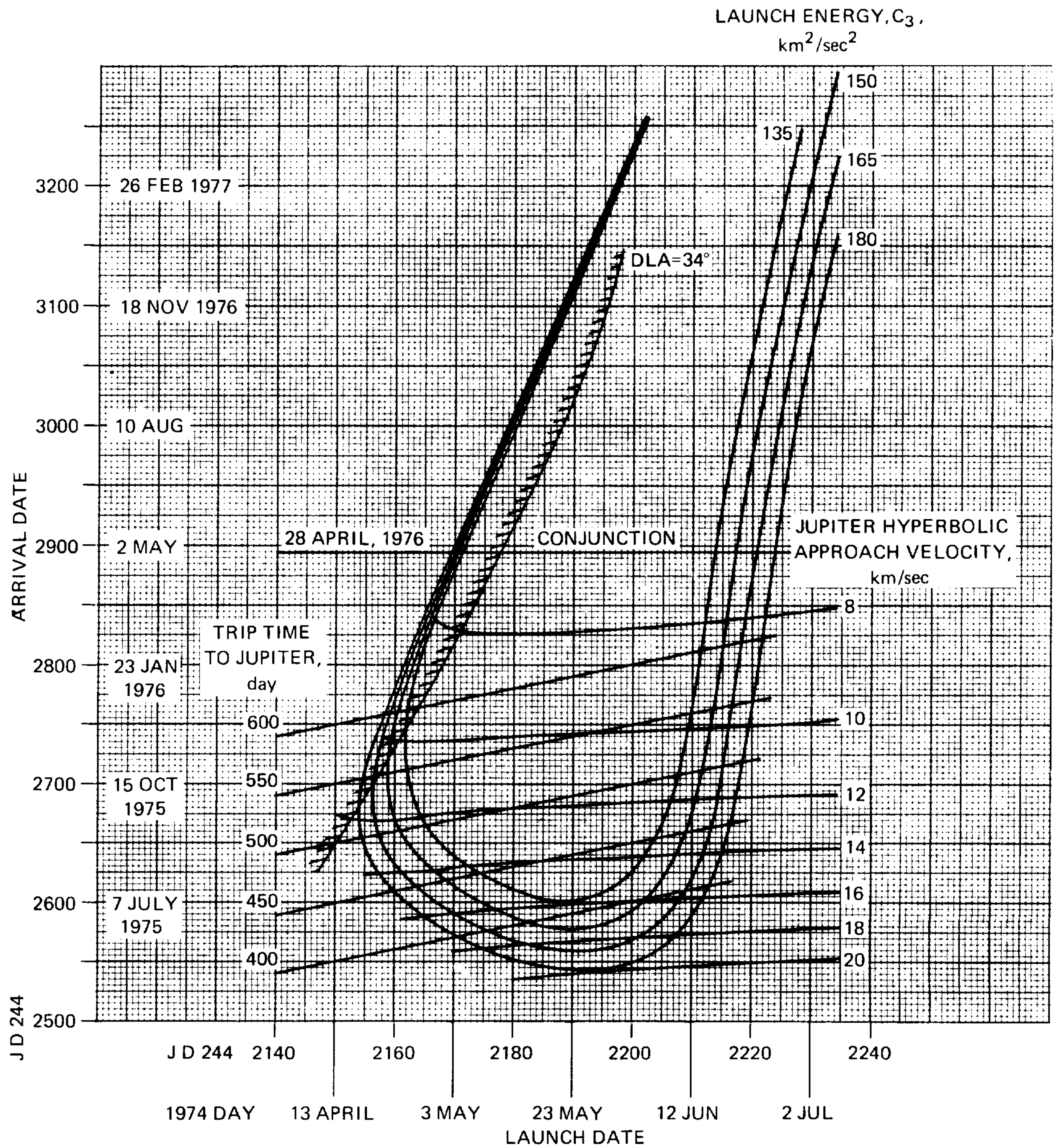

FIGURE 3-7. – JUPITER LAUNCH OPPORTUNITY 1974

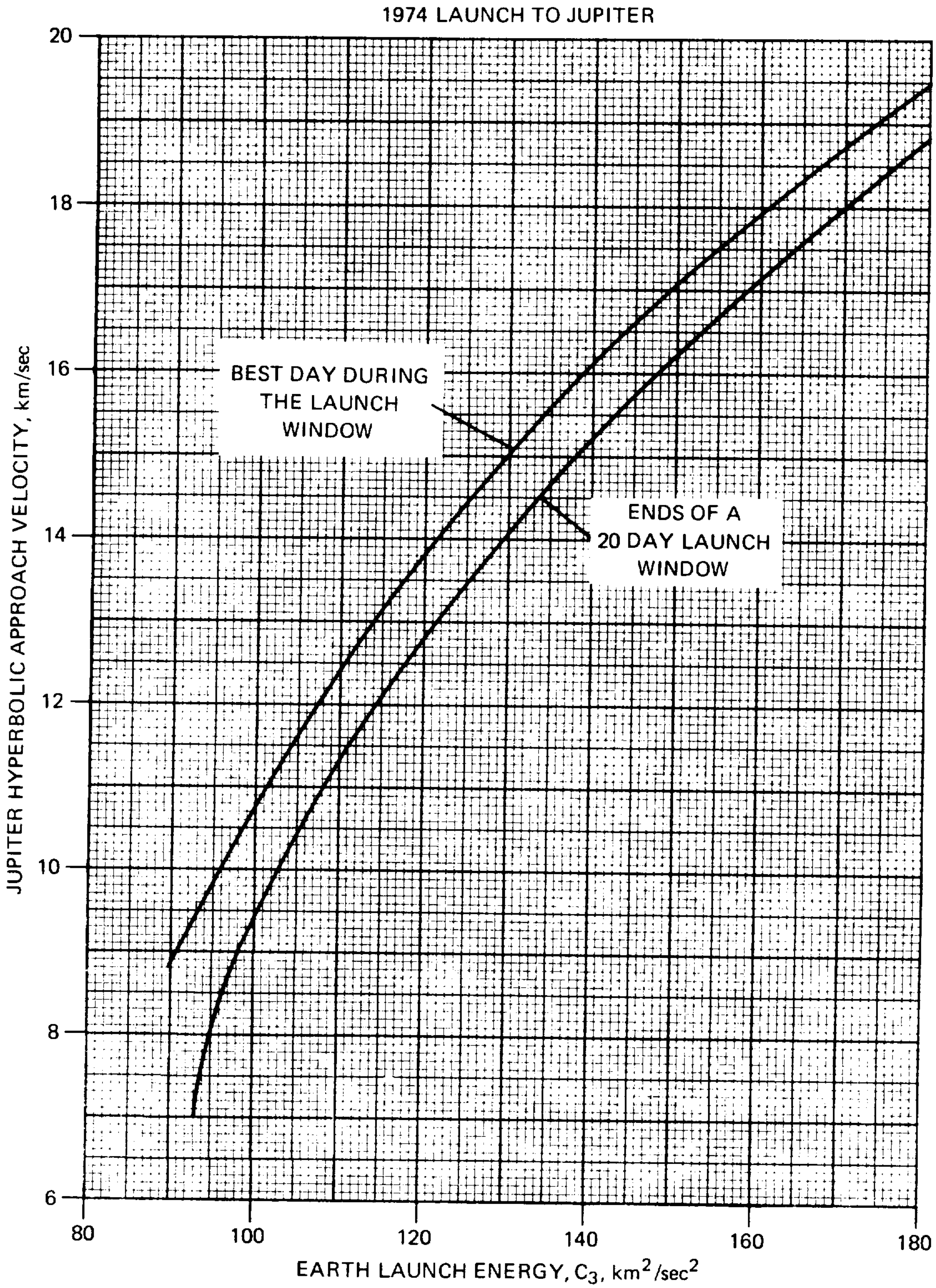

FIGURE 3-8.–LAUNCH ENERGY – APPROACH VELOCITY

3.3 TITAN/CENTAUR/TE 364-4 LAUNCH VEHICLE

The Titan 3D (5 segment solid motor) + D-IT Centaur + TE 364-4 launch vehicle provides a comfortable match with the trajectory requirements.

The launch energy available is shown in figure 3-9. The weights shown include the weight of the spacecraft to TE 364-4 adapter. The adapter used for Pioneer F weighs 15 pounds. The data has been calculated by the NASA Lewis Research Center.

For the Pioneer prototype spacecraft, the launch weight = 555 pounds plus 15 pounds for the adapter, or 570 pounds total. The launch C_3 = 182.0 km^2/sec^2 at 90° launch azimuth.

At 110° launch azimuth, the C_3 is reduced to 180 km^2/sec^2.

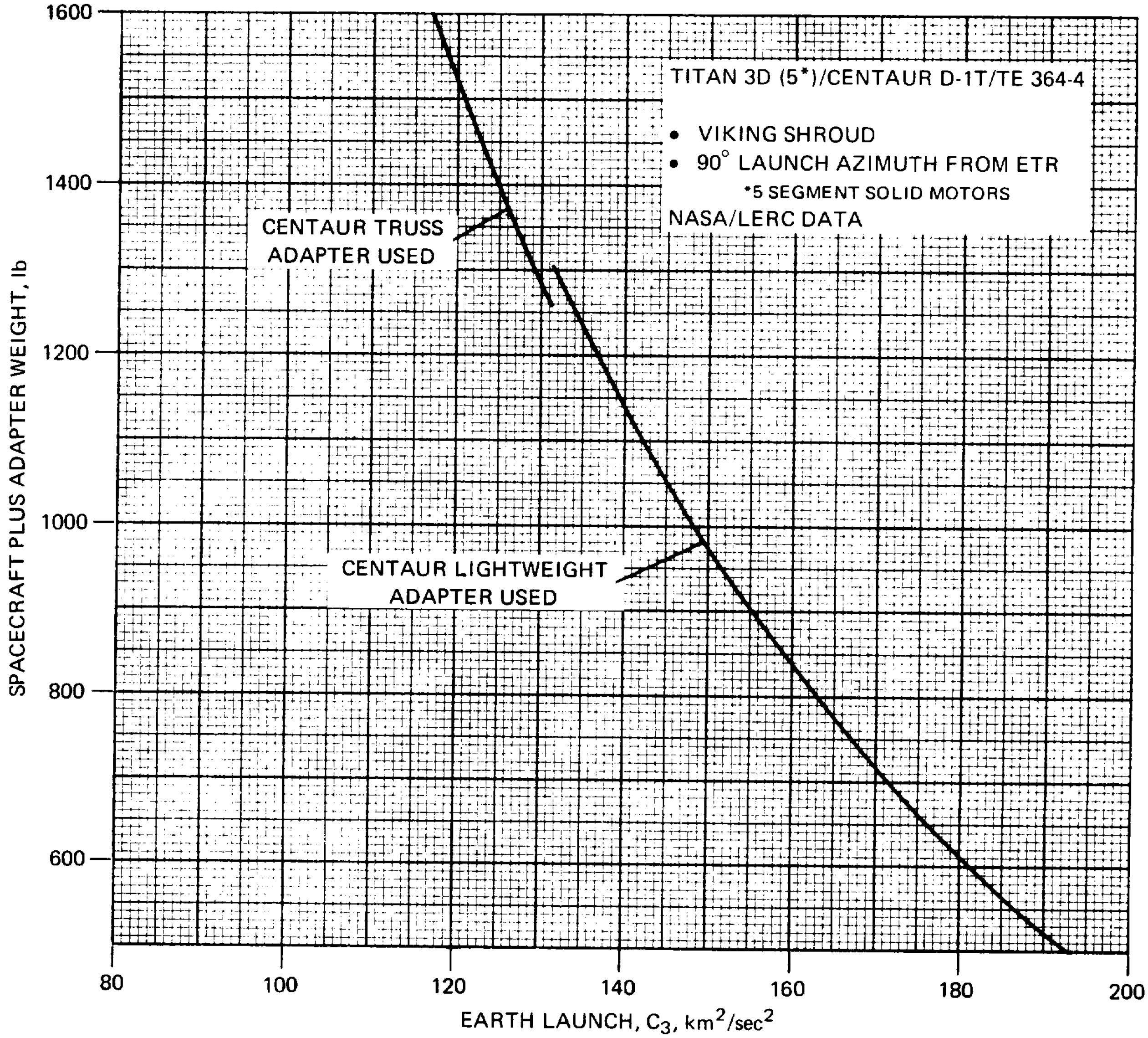

FIGURE 3-9. – LAUNCH VEHICLE PERFORMANCE

3.4 TARGETING AT JUPITER AND TRAJECTORY TRADE-OFFS - TITAN/CENTAUR

Mission Analysis leading to the choice of Nominal and Alternate Missions is described in section 3.1.

The Nominal Mission provides a 1.2 AU perhelion, 92.5° orbit inclination, and 2.2 years from swingby to the next ecliptic crossing. From figure 3-5 this mission requires a Jupiter hyperbolic approach velocity of 16.65 km/sec.

The Alternate Mission provides a 1.2 AU perihelion, 92.5° orbit inclination and 3.2 years from swingby to the next ecliptic crossing. From figure 3-5, the mission requires a Jupiter hyperbolic approach velocity of 16.52 km/sec.

Since the Jupiter approach velocities are approximately equal, both missions will have similar target requirements. Figure 3-10 shows a target map which applies to both the Nominal Mission and the Alternate Mission for a launch during the center of the Launch Window and 396 day trip to Jupiter.

The target map is constructed in a plane perpendicular to the Jupiter hyperbolic approach velocity. The T axis is parallel to the ecliptic. The small figure at the origin shows the size of the planet. The circles of closest approach distance show aim points in the target plane before the trajectories are deflected by Jupiter's gravity field. The size of the Jupiter impact circle compared to the actual size of the planet indicates the magnitude of Jupiter's deflection of the swingby trajectory.

The loci of inclination, perihelion distance, and time to ecliptic crossing show aim points which result in post swingby trajectories with the parameter values shown.

Targeting for θ angles between 90° and 180° produces a swingby trajectory which swings up out of the ecliptic and passes over the North Sun Pole before crossing the ecliptic. Targeting for θ angles between 180° and 270° produces a post-swingby trajectory which swings down out of the ecliptic and passes over the South Sun Pole before crossing the ecliptic.

The Nominal Mission is targeted for θ = 149.3°, and RCA = 3.2 Jupiter Radii. The Earth will be occulted for a short time during swingby.

The Alternate Mission is targeted for θ = 146.5° and RCA = 5.4 Jupiter Radii. The Earth will not be occulted.

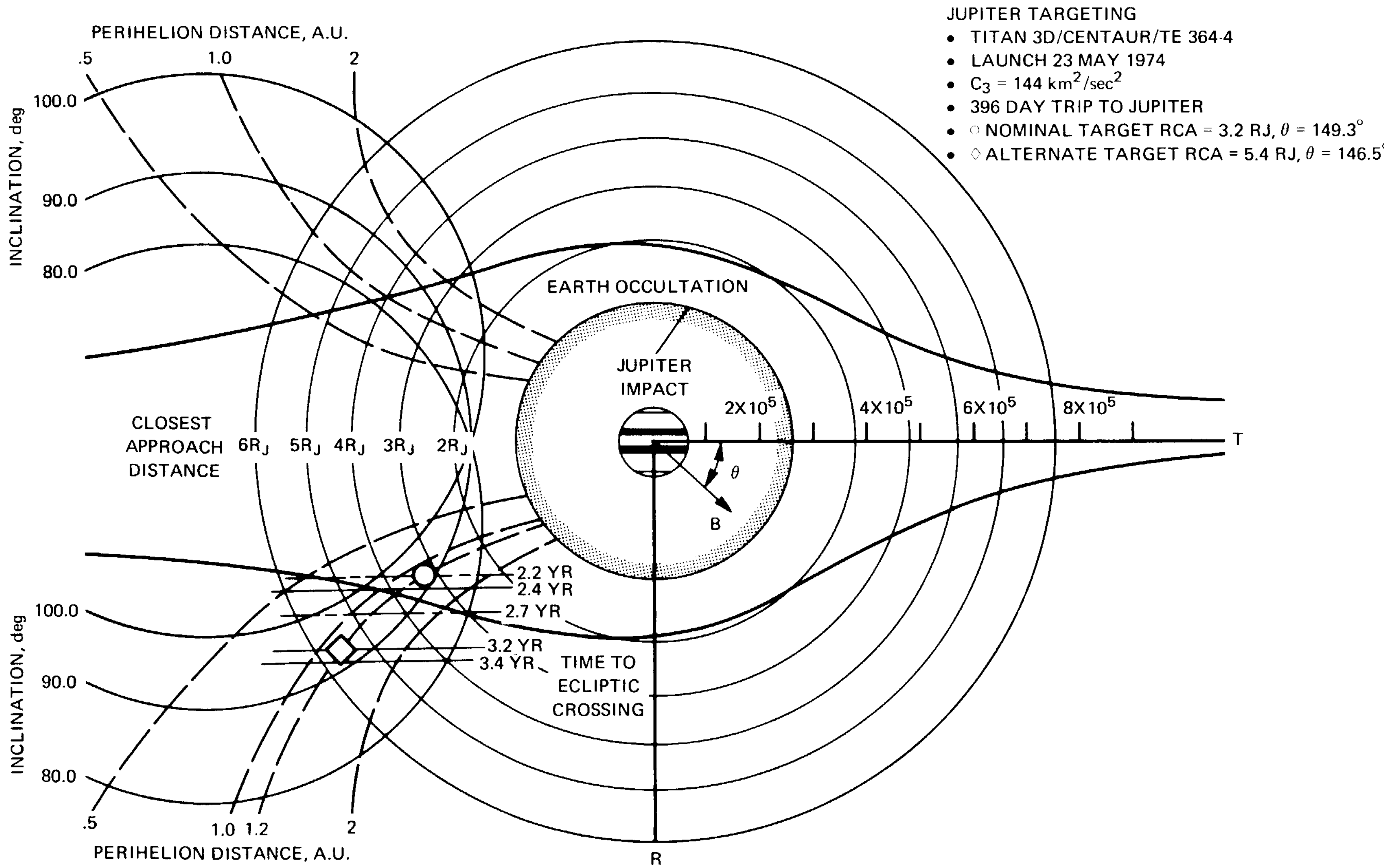

FIGURE 3-10. – JUPITER TARGET MAP

For launches at other times during the launch window and trip times to provide the same Jupiter hyperbolic approach velocity, the target map will be essentially the same as figure 3-10.

Trajectory trade-offs which can be performed with midcourse maneuvers and are possible with the same Jupiter hyperbolic approach velocity are found by examining figure 3-10. The post swingby inclination, perihelion distance, and time to ecliptic crossing are all related. Choosing any two determines the third, plus swingby RCA and θ.

Shorter trip times require closer passages to Jupiter.

Within the range of good trip times (2.2 - 2.4 years and 3.2 - 3.4 years) large changes in perihelion and RCA can be made with relatively small changes in post swingby inclination.

Earth to Jupiter trajectories which provide different values of Jupiter hyperbolic approach velocity allow different combinations of post swingby parameters with RCA and θ. The features of the target map will be similar, however to figure 3-10.

3.5 MISSION DESCRIPTION - TITAN NOMINAL

3.5.1 Launch Window

The launch window is determined by the requirement for Jupiter hyperbolic approach velocity = 16.65 km/sec. Trajectories from Earth to Jupiter will be chosen to provide a constant Jupiter hyperbolic approach velocity on each day within the boundary of launch energy available.

From figure 3-7, for Jupiter hyperbolic approach velocity = 16.65 km/sec, the launch window is:

begin date	JD 2442168	1 May 1974
close date	JD 2442211	13 June 1974
duration	43 days	

The launch window shortens by one day for every 20 pounds increase in launch weight.

The launch trajectory will include a parking orbit coast. For launch azimuths between 90° and 110° (the Pioneer F limits) the coast durations will be between 10 minutes and 19 minutes. The Centaur is currently qualified for up to 30 minutes coast duration.

The daily launch window for a launch azimuth range of 90° to 110° will be about two hours.

3.5.2 Trajectory to Jupiter

The flight times to Jupiter are chosen to provide a constant Jupiter approach velocity. At the opening of the launch window, flight time is 411 days. At the end of the window, flight time is 381 days.

Figures 3-11 and 3-12 show parameters of a typical Earth-to-Jupiter trajectory in the center of the launch window. The trajectory is almost a heliocentric hyperbola and would almost escape from the solar system without Jupiter encounter. The Earth to Jupiter trajectory plane is inclined less than 3 degrees to the ecliptic.

The Earth-to-Jupiter trajectory is similar to the Pioneer F Trajectory but requires less flight time. Canopus will be used as the roll reference during all of the flight.

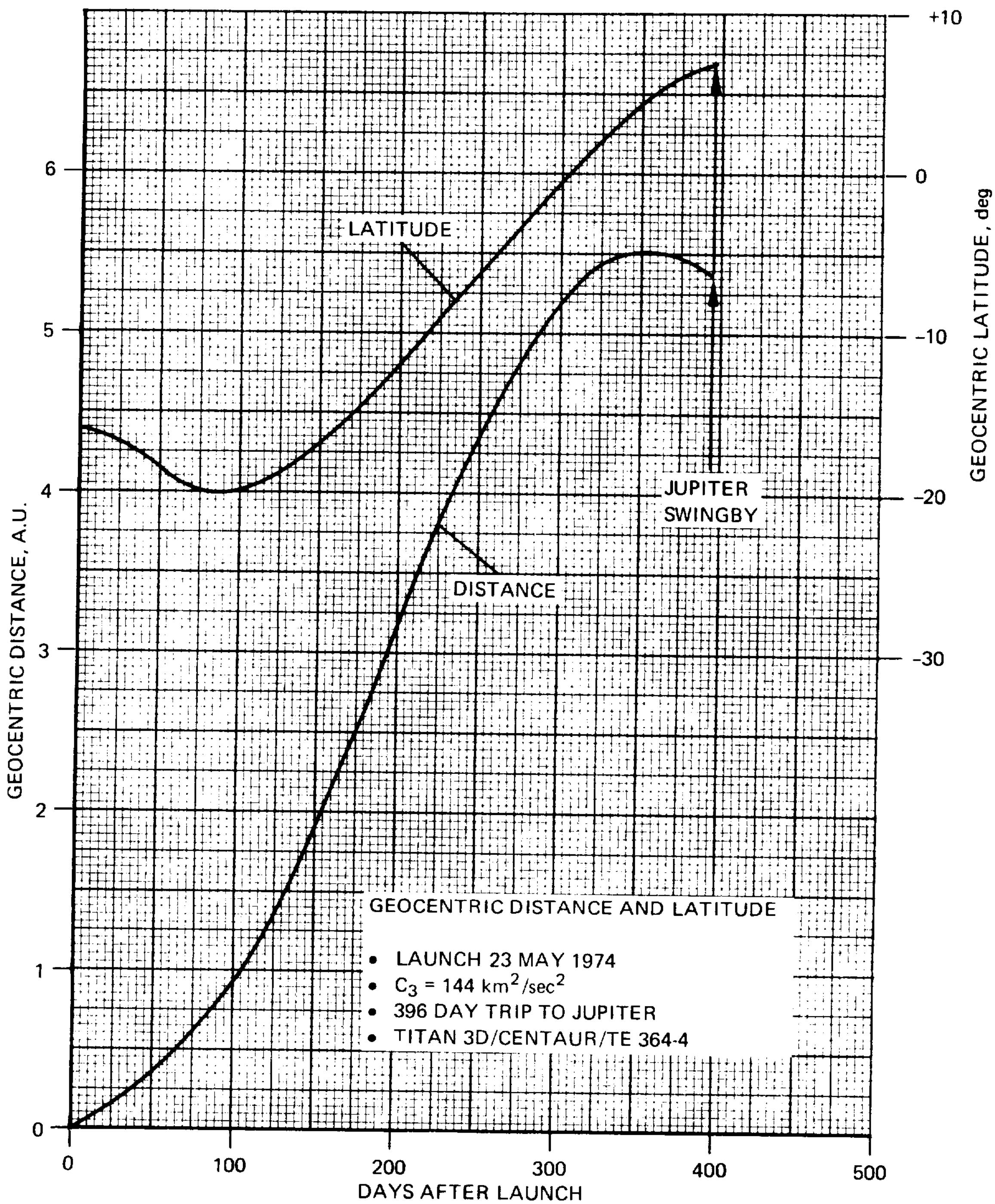

FIGURE 3-11. – EARTH TO JUPITER TRAJECTORY

HELIOCENTRIC DISTANCE AND REFERENCE ANGLES

- LAUNCH 23 MAY 1974
- $C_3 = 144\ km^2/sec^2$
- 396 DAY TRIP TO JUPITER
- TITAN 3D/CENTAUR/TE 364-4

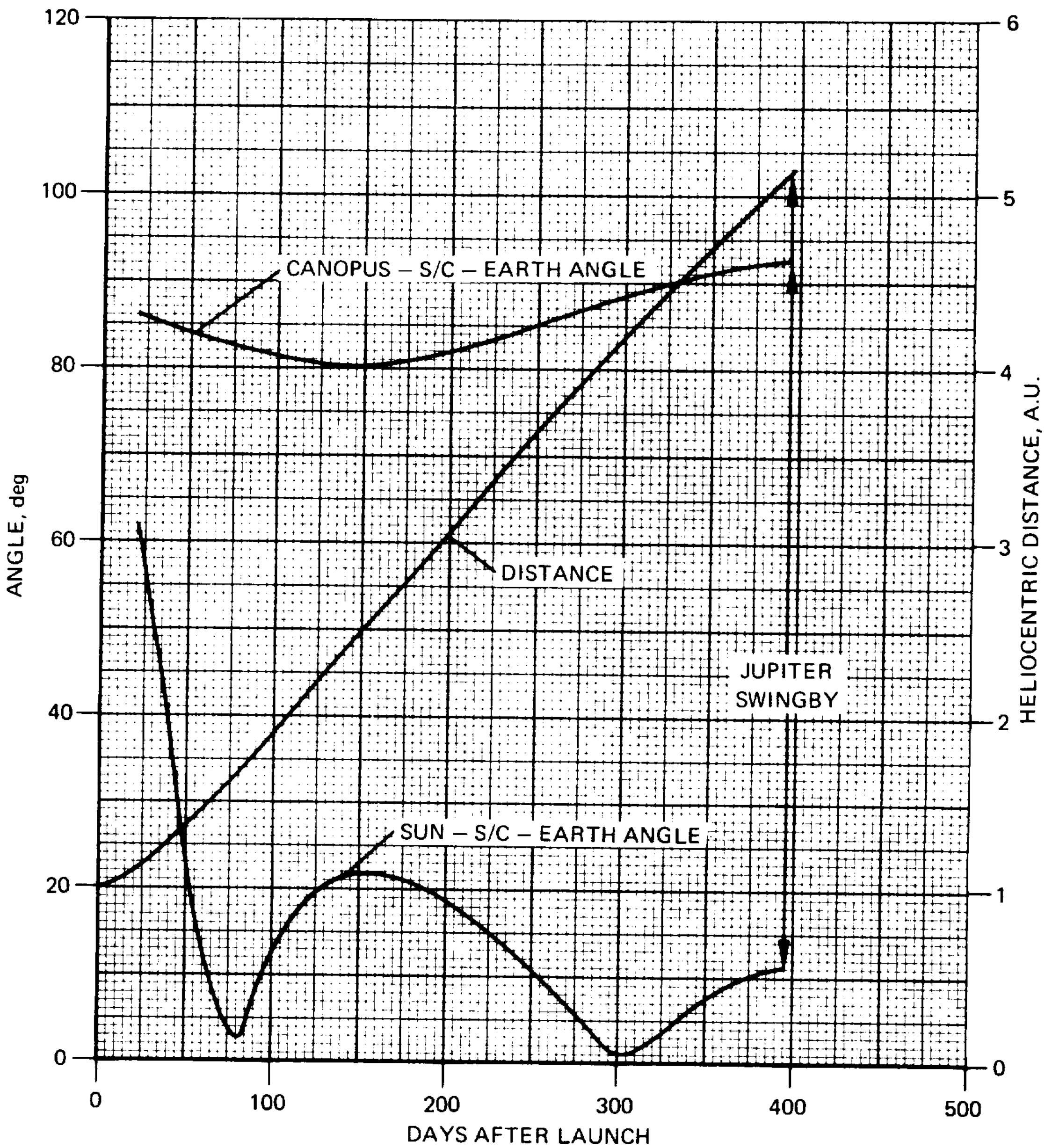

FIGURE 3-12. – EARTH TO JUPITER TRAJECTORY

3.5.3 Jupiter Swingby

The Jupiter Swingby trajectory must be chosen to match the approach velocity to the desired post encounter velocity. The approach velocity will be 16.65 km/sec for any arrival date. Since the direction of the approach velocity changes little with arrival data, the target conditions will be almost the same for any arrival data.

The target point is shown in figure 3-10.

The Earth will be occulted for about 38 minutes beginning 17 minutes after closest approach.

The spacecraft will pass through Jupiter's equator plane during departure, at about 3.9 Jupiter radii from Jupiter's mass center. It will pass at least one Jupiter radius from Almalthea (orbit radius = 2.5 R_J) and Io (orbit radius = 5.9 R_J).

The radiation environment during flyby is discussed in section 4.3.1.3 "Jupiter Radiation Belts".

Experiment visibility of Jupiter is discussed in section 5.4 "Experiment Viewing Requirements."

3.5.4 Midcourse Maneuvers and Target Accuracy

The midcourse maneuver strategy will be similar to Pioneer F.

The targeting accuracy required is ±1-1/2 days from nominal arrival time and within 12,000 km in the target plane.

These accuracies are within the capability of the Pioneer F system.

3.5.5 Postencounter Trajectory

The postencounter trajectory is an ellipse with perhelion = 1.2 A.U. The plane of the ellipse is tilted 92.5° to the plane of the ecliptic and the trajectory passes over the North Pole of the Sun before crossing the ecliptic.

Figure 3-13 shows the spacecraft latitude and distance after encounter as referenced to the Earth. Comparing the spacecraft latitude to the coverage available from the DSM 210' antenna stations (figure 3.3) shows that 24 hours per day coverage is available

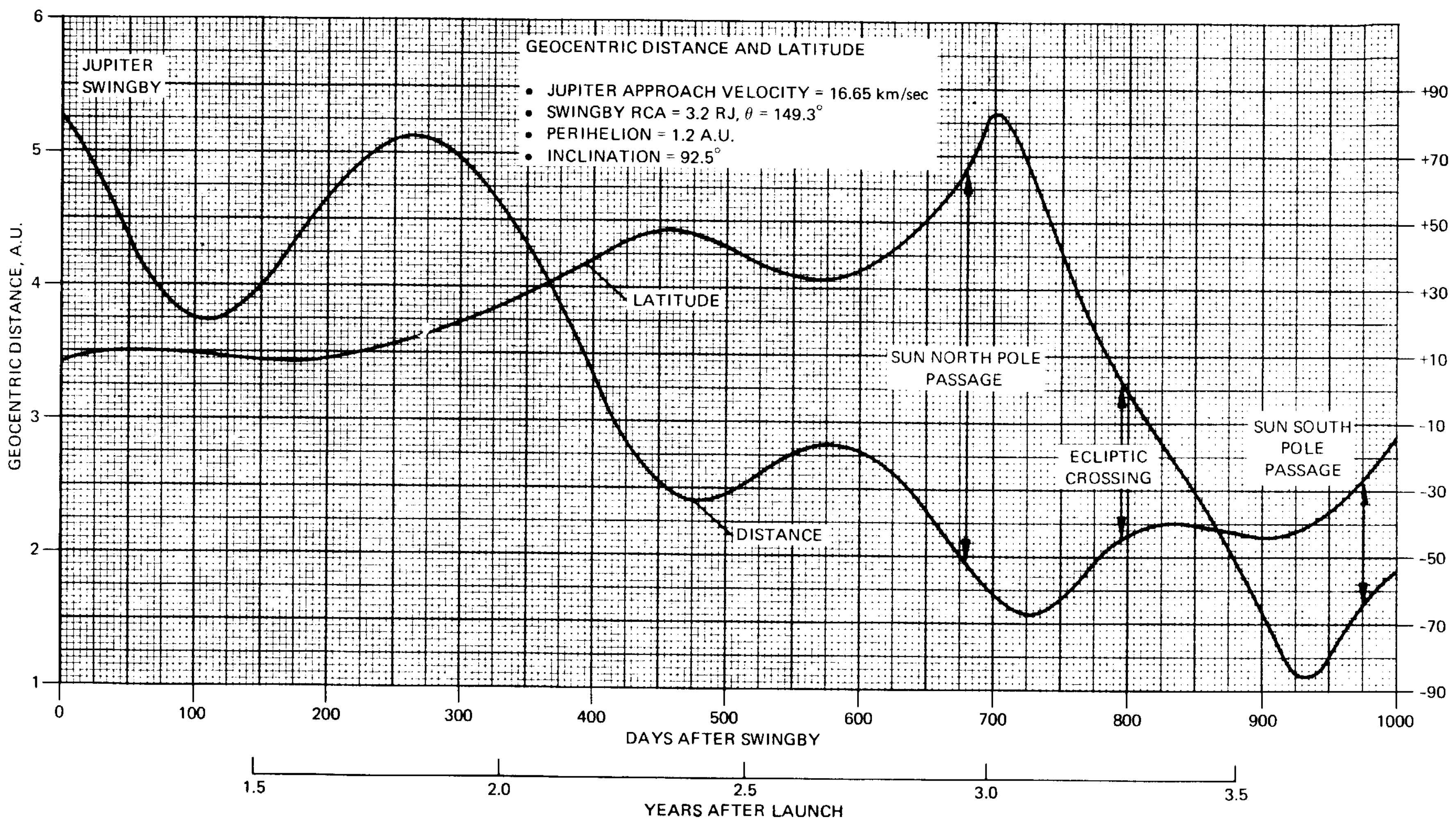

FIGURE 3-13.–NOMINAL OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

until 845 days after swingby. This is well after ecliptic crossing and provides good telemetry reception opportunities during the entire time the spacecraft above the ecliptic plane. Continuous coverage from 210' antennas is again available after day 895 and continues until after the South Sun Pole is passed.

Telemetry bit rates available are discussed in section 4.3.7, "Communications Subsystem".

Figure 3-14 shows the spacecraft latitude and distance after encounter as referenced to the Sun. The North Pole of the Sun is crossed 680 days after swingby at a distance of 1.7 A.U. Perihelion of 1.2 A.U. is passed about 775 days after swingby. The South Pole of the Sun is crossed 975 days after swingby at a distance of 2.4 A.U.

Figure 3-15 shows the spacecraft distance away from the ecliptic plane. The maximum distance of 2.1 A.U. occurs 560 days after swingby.

Figure 3-16 shows the time history of spacecraft attitude reference angles. The Pioneer F Canopus sensor has an angle range of 71° to 109°. Canopus will be within the field of view of the sensor and usable as a roll reference until 355 days after swingby. There will be a period of 80 days centered on ecliptic crossing when Canopus will also be usable as a roll reference.

During the times when Canopus is not usable, the Sun is always at least 10° away from the spacecraft spin axis and will provide a good roll reference.

The Sun-Spacecraft-Earth angle remains below 40° at all times and allows good experiment viewing of the sun.

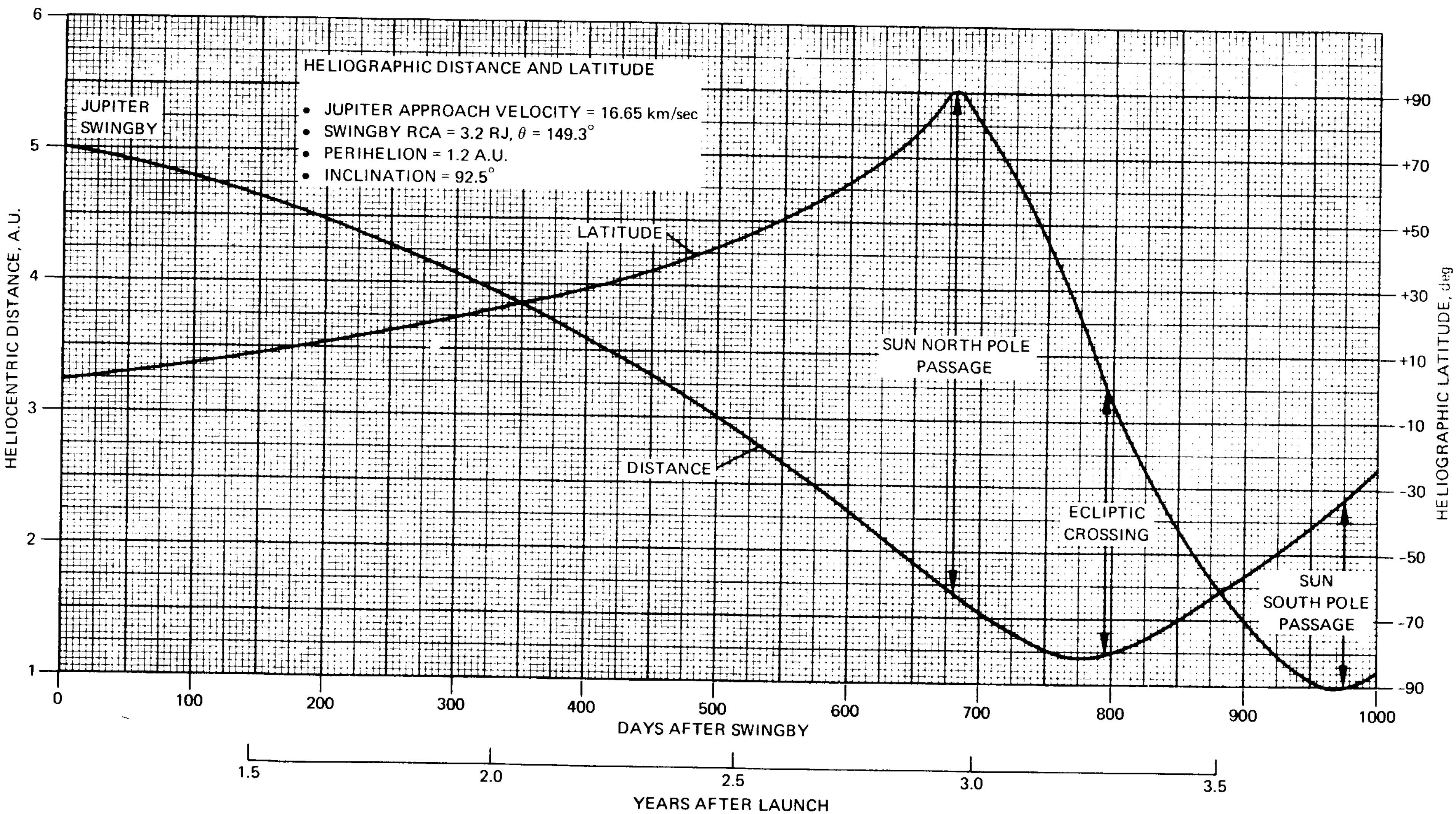

FIGURE 3-14.–NOMINAL OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

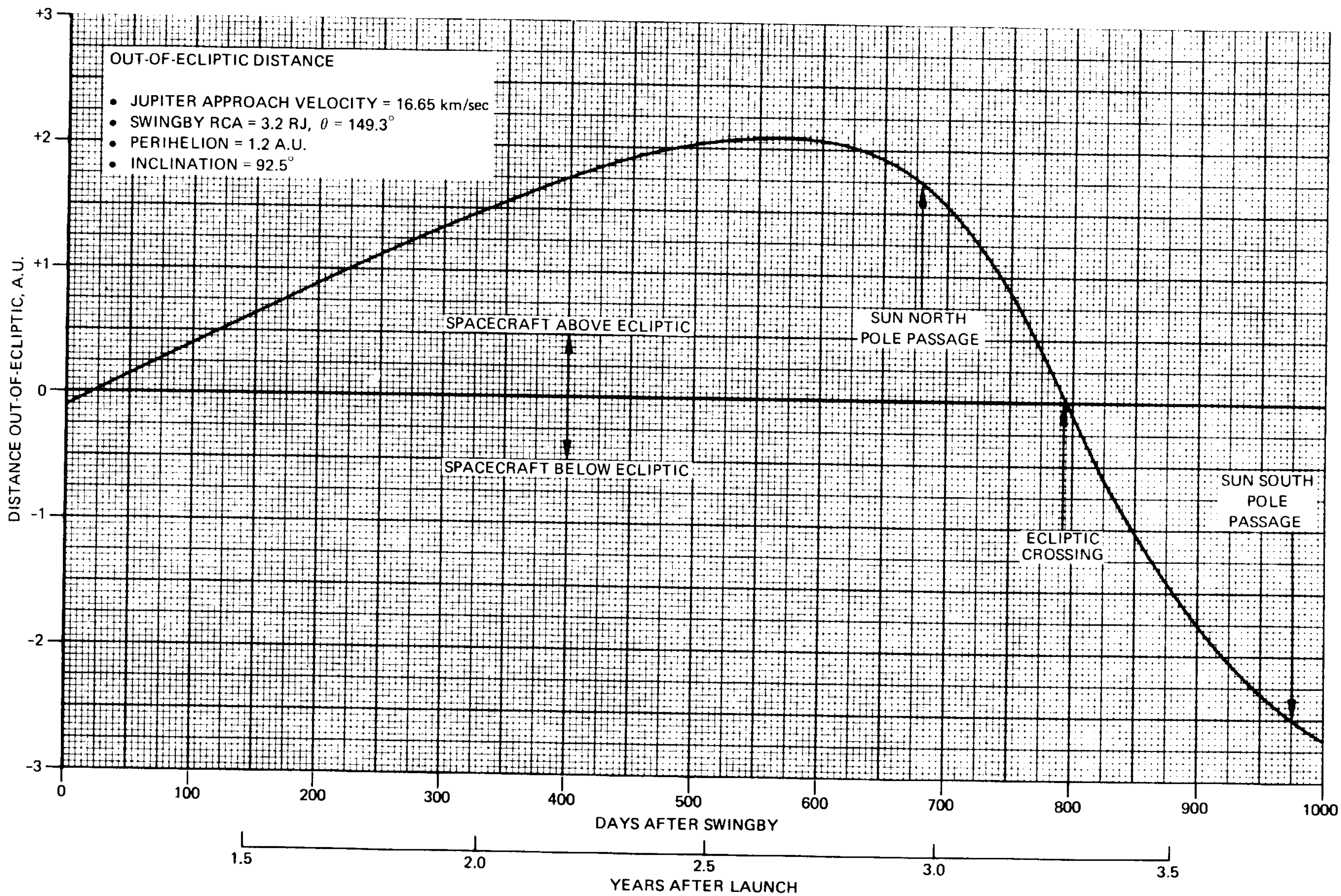

FIGURE 3-15.–NOMINAL OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

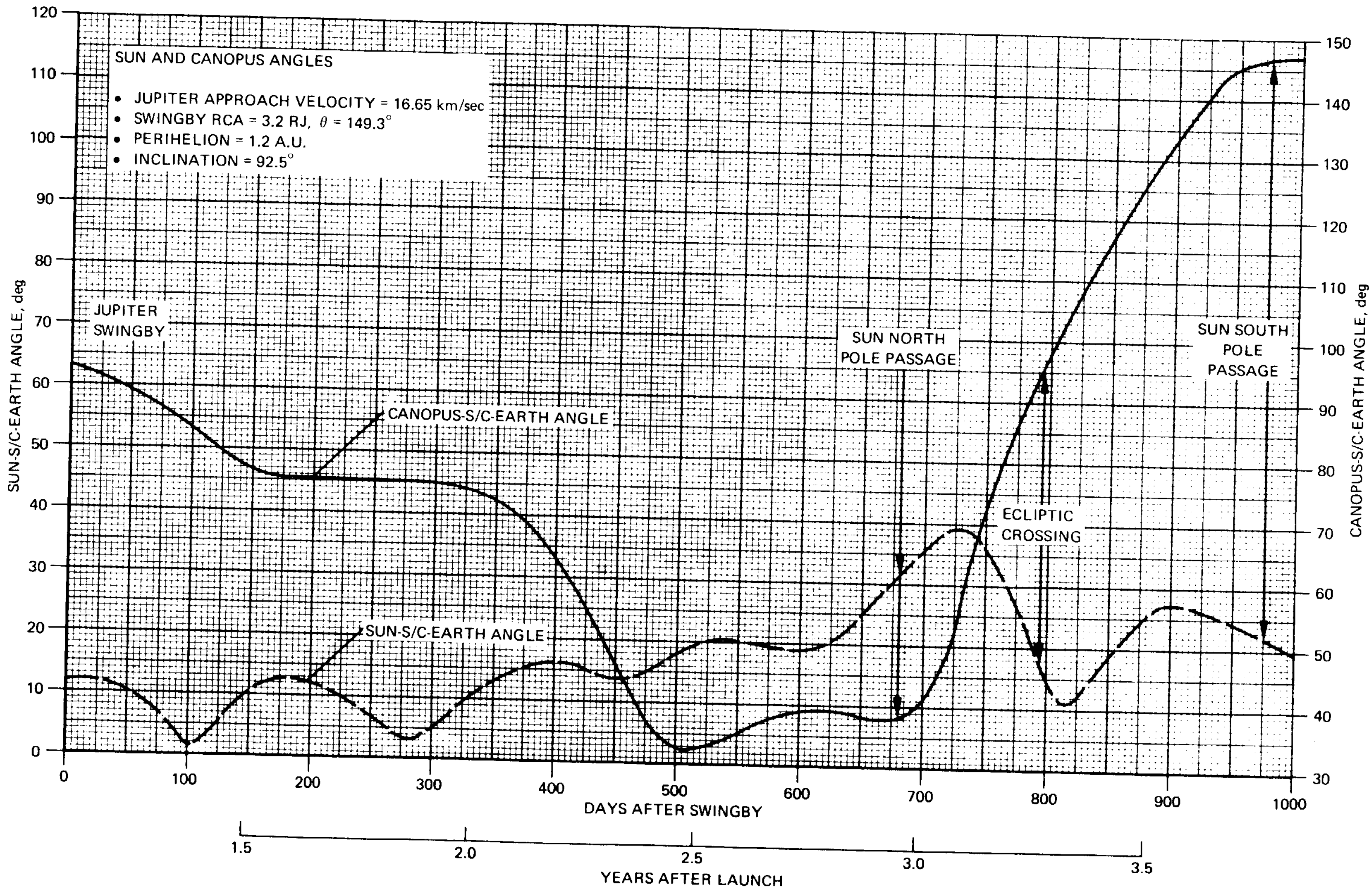

FIGURE 3-16.—NOMINAL OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

3.6 MISSION DESCRIPTION - TITAN ALTERNATE

3.6.1 Launch Window

The launch window for the alternate mission is almost the same as for the nominal mission because the Jupiter approach velocities are almost the same. Trajectories from Earth to Jupiter will be chosen to provide a Jupiter hyperbolic approach velocity = 16.52 km/sec on each day of the launch window.

From figure 3-7, the launch window is:

begin date	JD 2442167	30 April 1974
close date	JD 2442211	13 June 1975
duration	44 days	

The launch trajectory coast durations and daily launch window will be similar to the prime mission.

3.6.2 Trajectory to Jupiter

Flight times to Jupiter are slightly longer than the nominal mission. At the opening of the window, the flight time is 413 days. At the end of the launch window, flight time is 383 days.

The trajectory parameters shown in figures 3-11 and 3-12 describe the Alternate Mission Flight to Jupiter.

3.6.3 Jupiter Swingby

The Jupiter Swingby trajectory must be chosen to match the approach velocity to the departure velocity. The target point is shown in figure 3-10.

There will be no Earth occultation.

The spacecraft will pass through Jupiter's equator plane during departure, at about 7.5 Jupiter radii from Jupiter's mass center. It will pass at least 1.5 Jupiter radii from Io (orbit radius = 5.9 R_J) and Europa (orbit radius = 9.4 R_J).

The radiation environment during flyby is discussed in section 4.3.1.3 "Jupiter Radiation Belts".

Experiment visibility of Jupiter is discussed in section 5.4 "Experiment Viewing Requirements".

3.6.4 Midcourse Maneuvers and Target Accuracy

The midcourse maneuver strategy will be similar to Pioneer F.

The targeting accuracies required are the same as the nominal mission and are within the capability of the Pioneer F system.

The Alternate Mission can be retargeted from a Nominal Mission launch by a midcourse maneuver that is within the capability of the Pioneer F system.

3.6.5 Post Encounter Trajectory

The postencounter trajectory is an ellipse with perihelion = 1.2 A.U. The plane of the ellipse is tilted 92.5° to the plane of the ecliptic and the trajectory passes over the North Pole of the Sun before crossing the ecliptic. The Alternate Mission will required about one year longer than the Nominal Mission to return to the Sun.

Figure 3-17 shows the spacecraft latitude and distance as seen from the Earth. Comparing the latitude with the coverage available from the DSM 210' antenna stations shows that 24 hour per day coverage is available until 1200 days after swingby. This is well after ecliptic crossing and provides good telemetry reception opportunities while the spacecraft is above the ecliptic plane. Continuous coverage is again available after day 1243 and continues until after the Sun South Pole is passed.

Telemetry bit rates are discussed in section 4.4.7, "Communications Subsystem".

Figure 3-18 shows the spacecraft latitude and distance as seen from the Sun. The spacecraft will pass aphelion about 125 days after swingby and will pass over the North Pole of the Sun 1000 days after swingby at a distance of 2.1 A.U. Perihelion of 1.2 A.U. is passed at 1153 days after swingby just after the spacecraft crosses the ecliptic plane. The South Pole of the Sun is crossed 1280 days after swingby at a distance of 1.85 A.U.

Figure 3-19 shows the spacecraft distance away from the ecliptic plane. The maximum distance of 2.7 A.U. occurs 800 days after swingby.

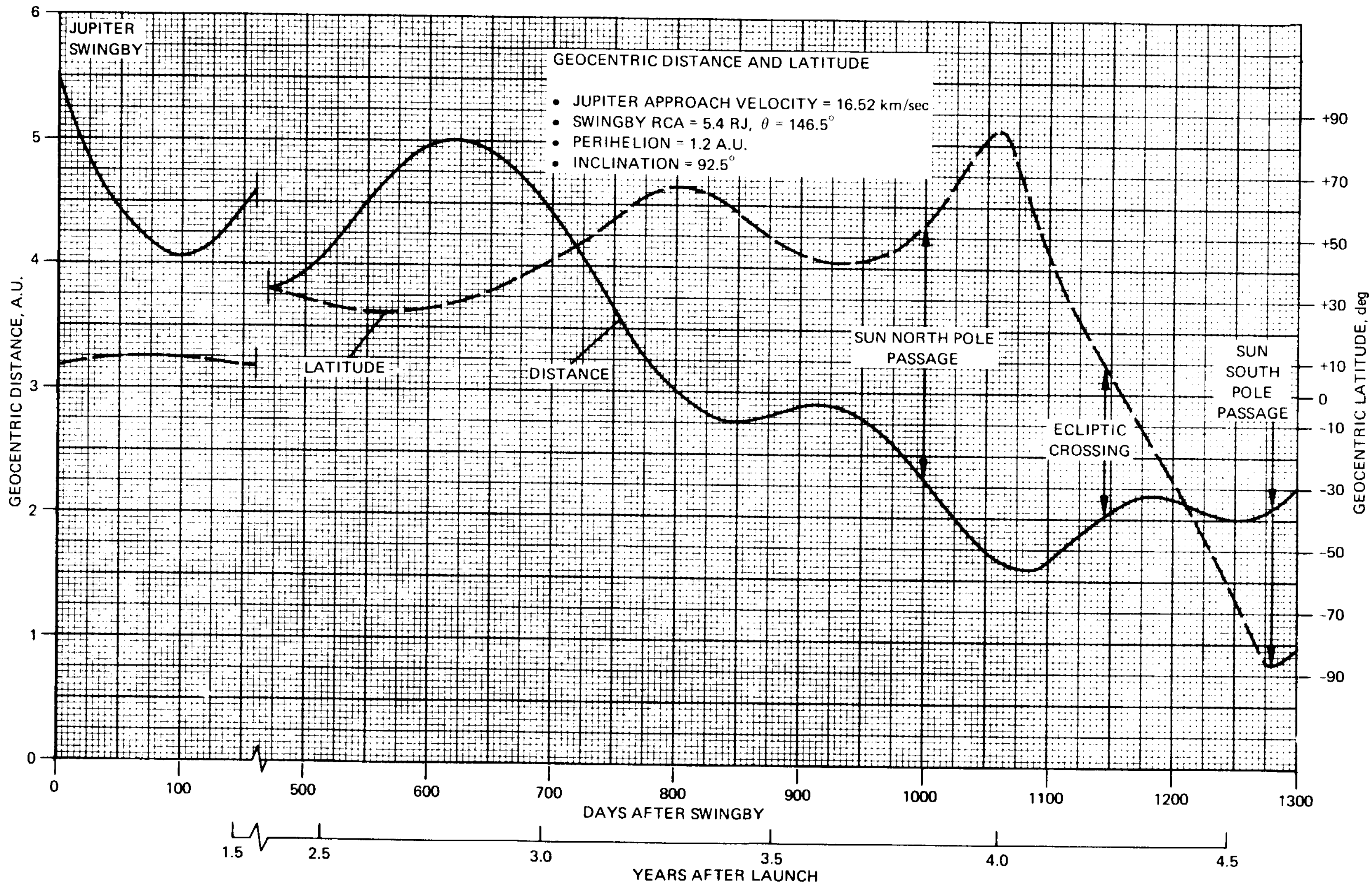

FIGURE 3-17.–ALTERNATE OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

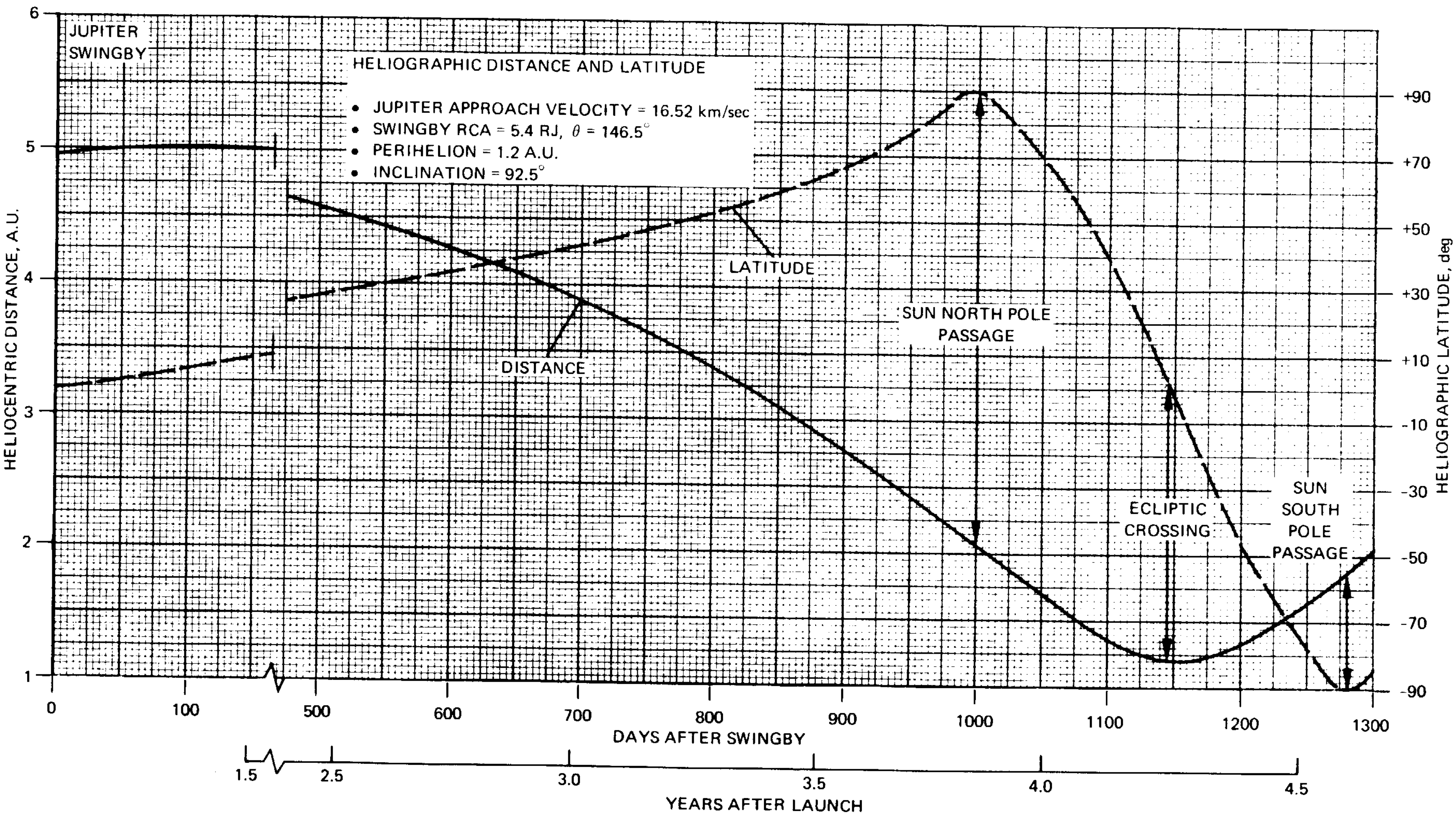

FIGURE 3-18.–ALTERNATE OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

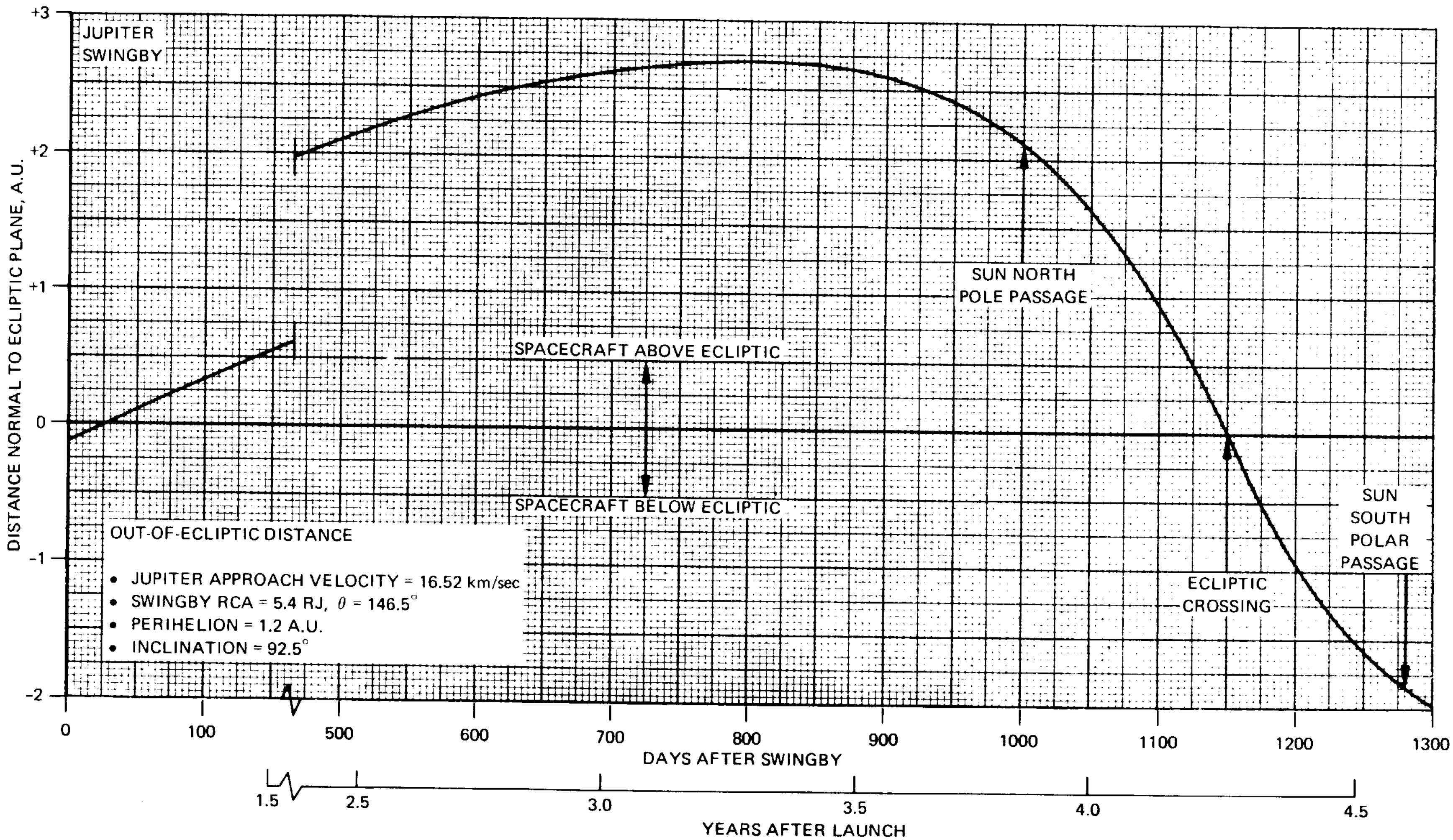

FIGURE 3-19.–ALTERNATE OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

Figure 3-20 shows the time history of attitude reference angles. Canopus will be within the field of view of the Pioneer F sensor and usable as a roll reference until 390 days after swingby. There will be a period of 77 days centered on ecliptic crossing when Canopus will also be usable as a roll reference.

During the times when Canopus is not usable as a roll reference, the Sun will be used. There are two time periods (around 460 days after swingby and around 640 days after swingby) when the Sun will be within 10° of the spacecraft spin axis and degrade the roll reference.

The Sun-Spacecraft-Earth angle remains below 40° at all times and allows good experiment viewing of the Sun.

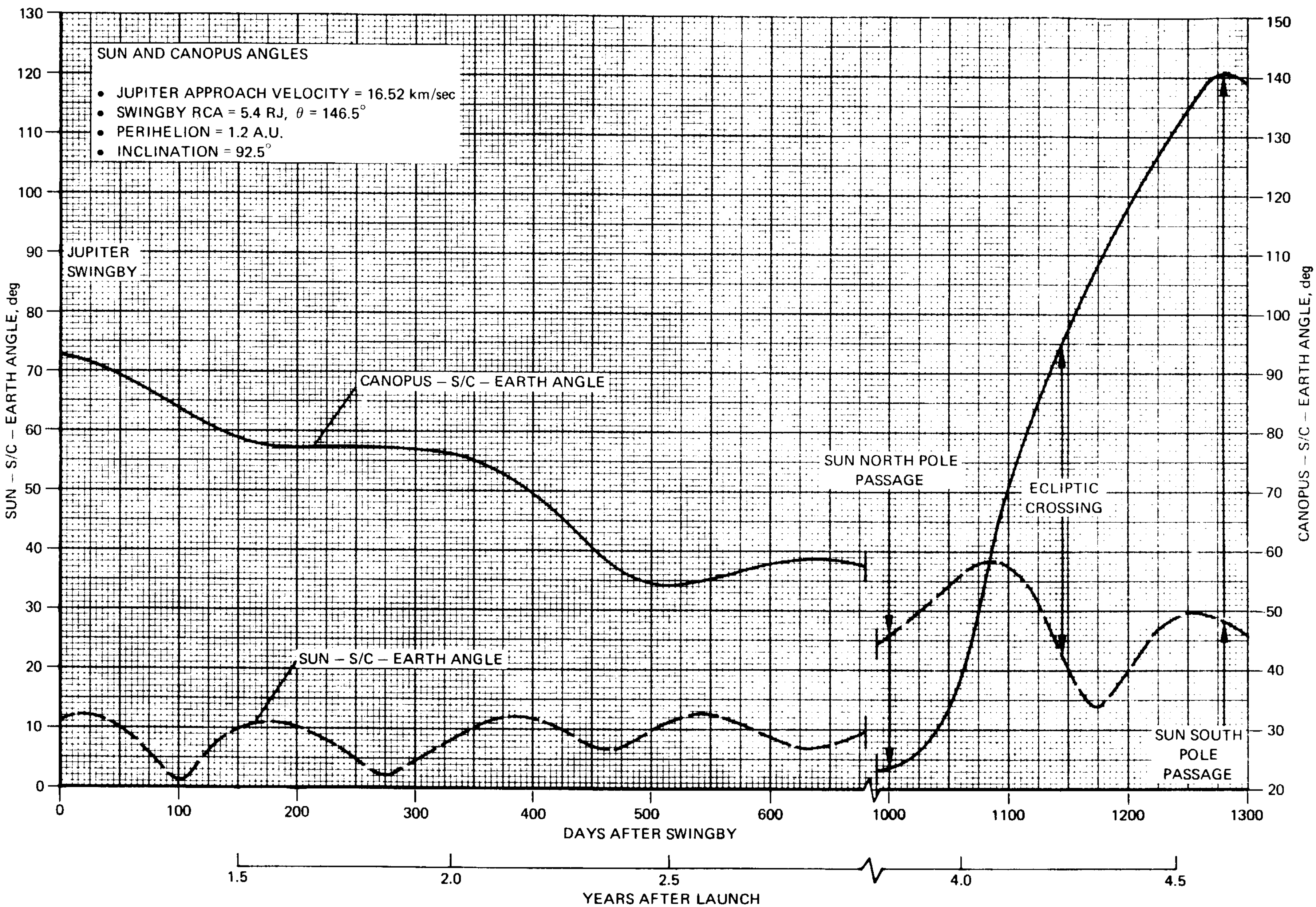

FIGURE 3-20.–ALTERNATE OUT-OF-ECLIPTIC TRAJECTORY TITAN 3D/CENTAUR/TE 364-4

3.7 ALTERNATE LAUNCH VEHICLES

3.7.1 Introduction

The Atlas/Centaur/TE 364-4 and Titan 3C/TE 364-4 have been investigated as launch vehicles for an out-of-ecliptic mission. Both of these launch vehicles provide less mission capability and flexibility than the Titan 3D/Centaur/TE 364-4, but have lower costs and may allow fewer conflicts for launch facility use.

3.7.2 Mission Comparisons

The Atlas/Centaur/TE 364-4 and Titan 3C/TE 364-4 launch vehicles both provide lower launch energies and lower Jupiter hyperbolic approach velocities than does the Titan 3D/Centaur/TE 364-4.

The Atlas/Centaur/TE 364-4 approach velocities are low enough that mission duration and heliocentric inclination change significantly throughout the launch window. The postswingby trajectories will return to the Sun "below" the Ecliptic Plane and pass over the South Pole of the Sun before crossing the Ecliptic. For the same inclination to the Ecliptic the Sun's tilt allows South polar passage first trajectories to travel over higher heliographic latitudes than North polar passage first trajectories.

The Titan 3C/TE 364-4 approach velocities are high enough to provide less variation in mission duration and inclination throughout the launch window. The post-swingby trajectories will return to the Sun "above" the Ecliptic Plane and pass over the North Pole of the Sun before crossing the Ecliptic. They will be visible from the Northern hemisphere of Earth for most of the mission and allow good coverage from the DSM 210' antennas.

Mission comparisons are shown in figures 3-21 and 3-22. The Atlas/Centaur/TE 364-4 provides missions with lower inclinations and longer trip times, than the Titan 3D/Centaur/TE 364-4. The Titan 3C/TE 364-4 provides missions which are very similar to those available from the Titan 3D/Centaur/TE 364-4.

POSTSWINGBY INCLINATION TO ECLIPTIC

- 1974 LAUNCH TO JUPITER
- 1.2 A.U. POSTSWINGBY PERIHELION
- MISSION DURATION PER FIG. 3-22

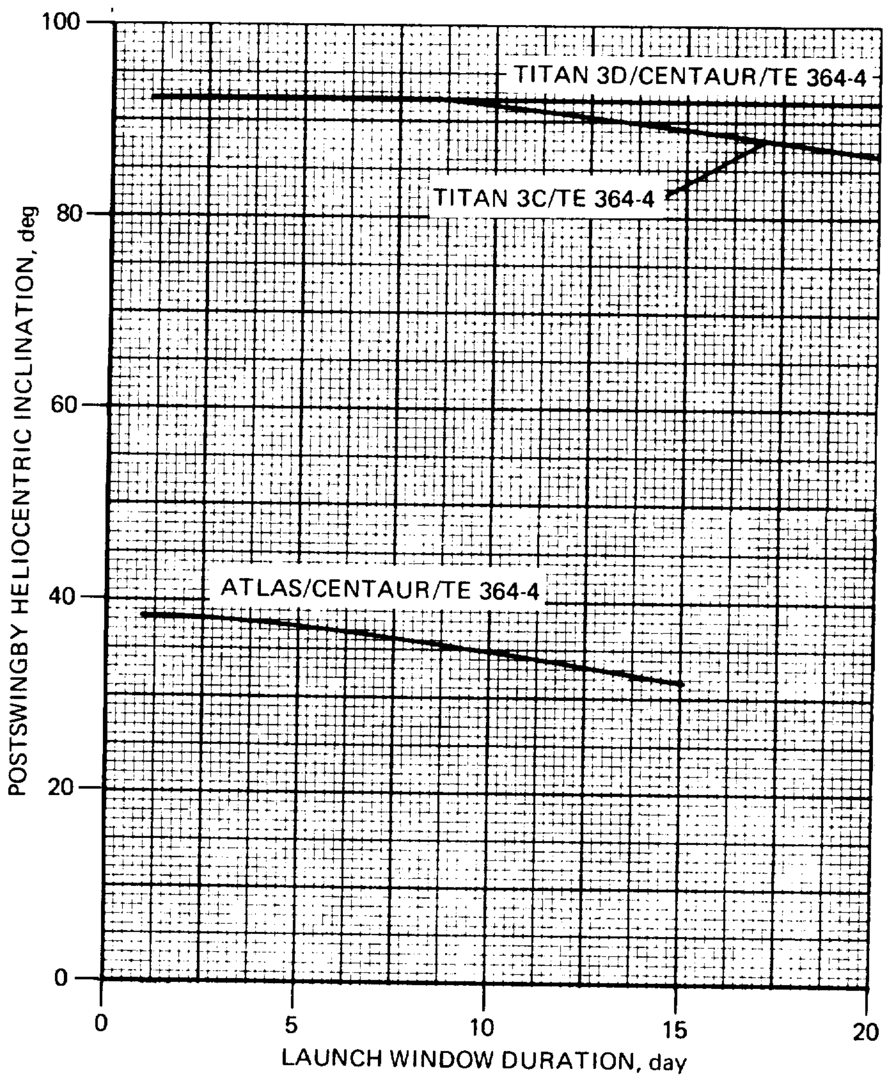

FIGURE 3-21.—LAUNCH VEHICLE COMPARISON

MISSION DURATIONS

- 1974 LAUNCH TO JUPITER
- 1.2 A.U. POSTENCOUNTER PERIHELION
- POSTSWINGBY INCLINATION PER FIG. 3-21
- ATLAS/CENTAUR PASSES SOUTH SOLAR POLE FIRST
- TITAN/CENTAUR PASSES NORTH SOLAR POLE FIRST

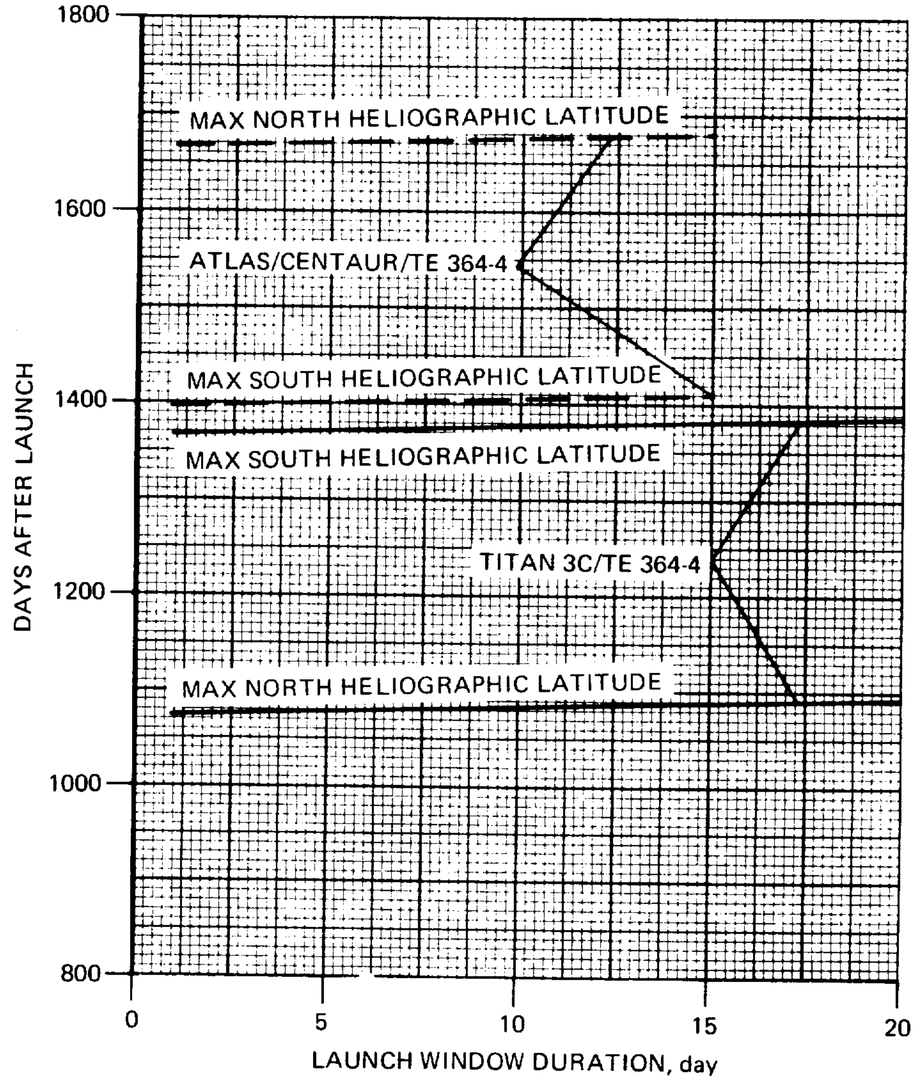

FIGURE 3-22.–LAUNCH VEHICLE COMPARISON

3.7.3 Atlas/Centaur/TE 364-4

3.7.3.1 Launch Energy

The Atlas/Centaur/TE 364-4 is the same launch vehicle used for Pioneer G. The launch energy available is plotted in figure 3-23 from data furnished by NASA Lewis Research Center. For the Pioneer spacecraft weight of 555 pounds, the launch $C_3 = 90.6$ km²/sec².

3.7.3.2 Launch Window

Figure 3-24 shows the 1974 Launch window for an Atlas/Centaur/TE 364-4.

For $C_3 = 90.6\ km^2/sec^2$, a 15 day launch window is:

	Date		Trip Time to Jupiter	Jupiter Hyperbolic Approach Velocity
open	JD2442177	10 May	640 days	8.0 km/sec
best day	JD2442184	17 May	594 days	8.9 km/sec
close	JD2442192	25 May	635 days	8.0 km/sec

3.7.3.3 Postswingby Trajectory Inclination

Figure 3-25 shows the range of trajectory plane inclinations which can be achieved with Atlas/Centaur/ TE 364-4 approach velocities. For these approach velocities, the perihelion distance influences the inclination which can be achieved.

A perihelion distance of 1.2 A.U. has been used for the nominal mission. The perihelion distance could be increased to about 1.5 A.U. to allow slightly higher inclinations without violating any experiment or spacecraft constraint.

Since the Atlas/Centaur/TE 364-4 does not allow missions which fly over the Sun poles, the highest Sun latitude depends on the trajectory plane orientation. The Sun's spin axis tilt allows trajectories which pass below the ecliptic to achieve a higher Sun latitude than trajectories which pass above the ecliptic plane. The highest Sun latitude will be 5° greater than the trajectory plane inclination for a 1974 launch.

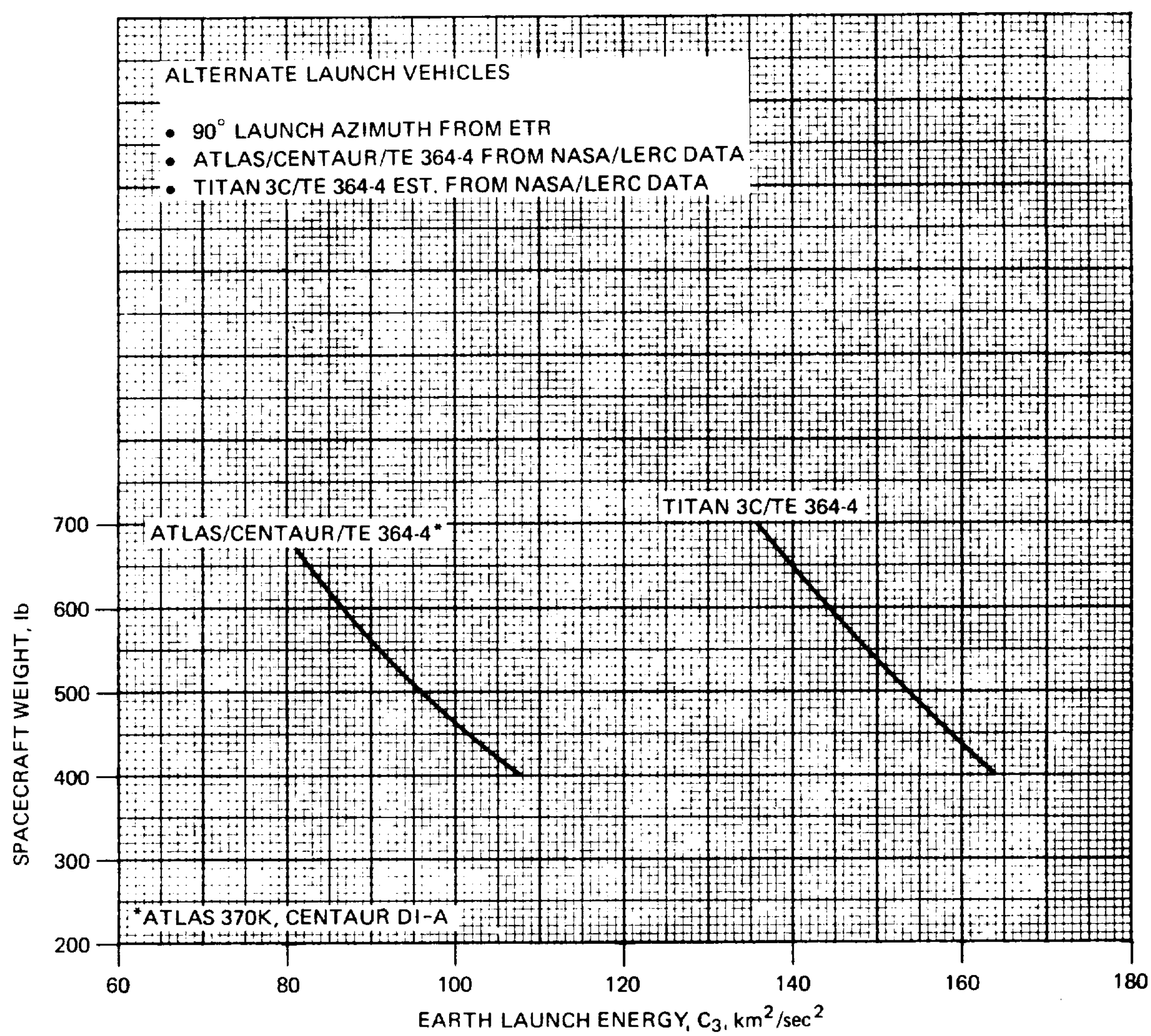

FIGURE 3-23.—ALTERNATE LAUNCH VEHICLE PERFORMANCE

1974 LAUNCH WINDOW TO JUPITER

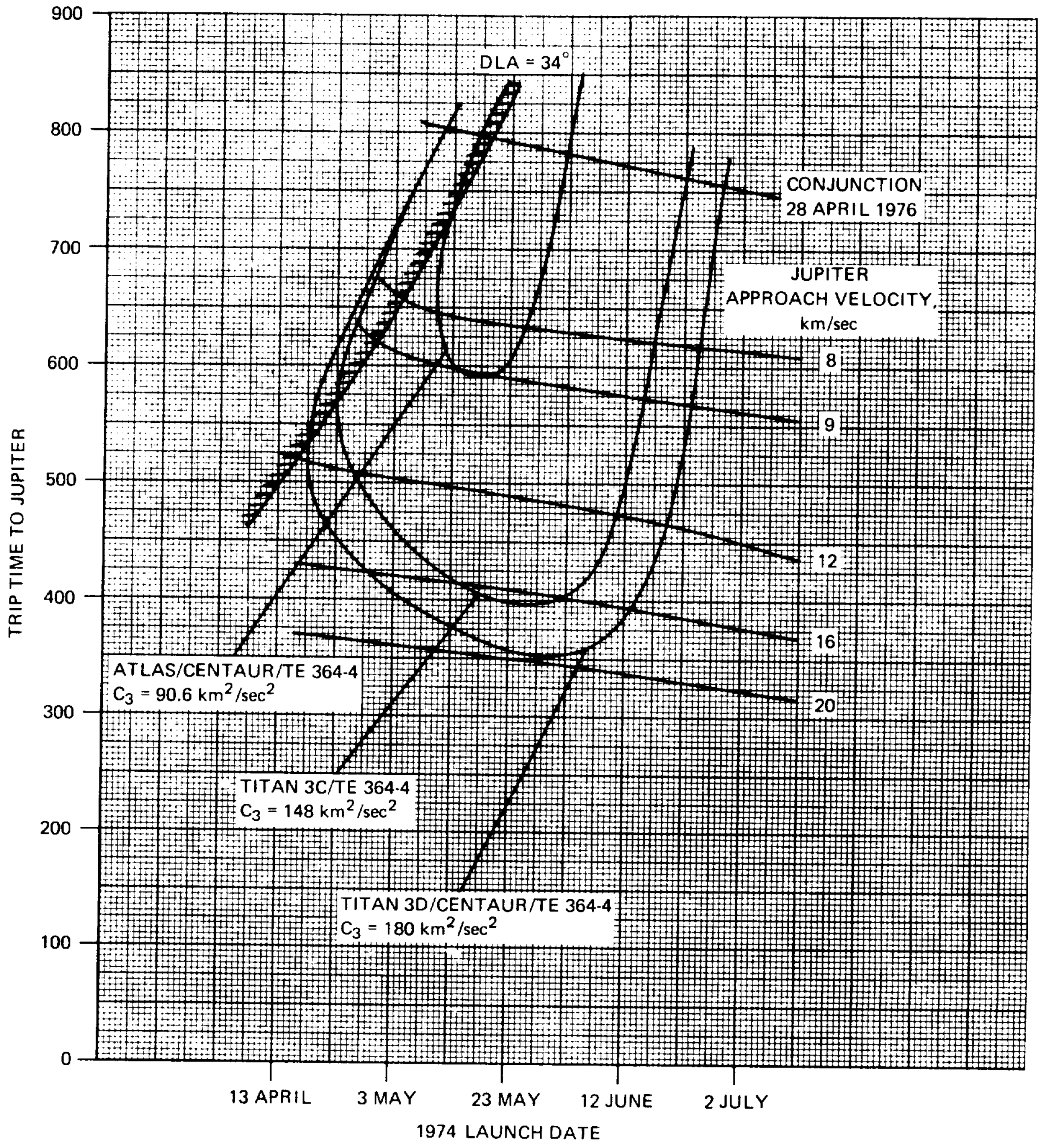

FIGURE 3-24.–LAUNCH VEHICLE COMPARISON

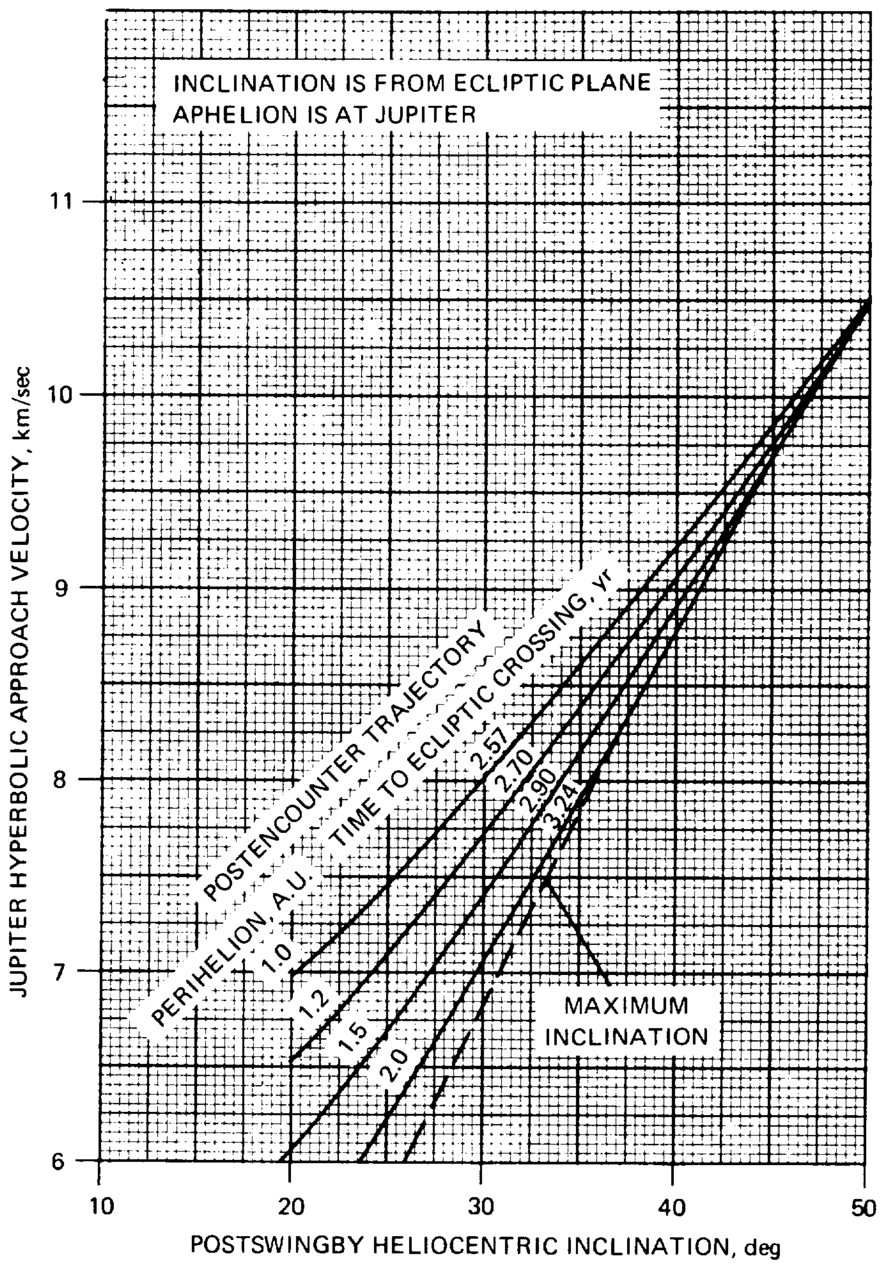

FIGURE 3-25.—POSTSWINGBY INCLINATION

3.7.3.4 Trip Time and Targeting

The trip time from swingby to ecliptic crossing must provide good Sun-Spacecraft-Earth angles.

Figures 3-26 shows the Earth position at swingby and the range of Earth positions which provide good Sun-Spacecraft-Earth angles. The trip times from swingby to ecliptic crossing are 0.6 - 1.0 years, 1.6 - 2.0 years, 2.6 - 3.0 years, and 3.6 - 4.0 years.

The trip time of 2.6 years from swingby to ecliptic crossing provides maximum inclination and has been chosen for the nominal Atlas/Centaur mission on the best day of the launch window.

The low Jupiter hyperbolic approach velocities require swingby farther from Jupiter than for the Titan missions.

A summary of the postswingby trajectory and targeting characteristics is:

Launch Day	Swingby RCA	Swingby θ	Postswingby inclination to Ecliptic	Postswingby perihelion	Time to ecliptic crossing
10 May 1974	$27R_J$	216°	31.5°	1.2 AU	2.45 yr
17 May 1974	$21R_J$	219°	38.5°	1.2 AU	2.60 yr
25 May 1974	$27R_J$	216°	31.5°	1.2 AU	2.45 yr

3.7.3.5 Nominal Mission - Atlas/Centaur

Figures 3-27 to 3-30 present information describing the nominal Atlas/Centaur trajectory on the best day of the launch window.

The trajectory passes below the ecliptic and reaches a maximum heliographic latitude of 43.5°.

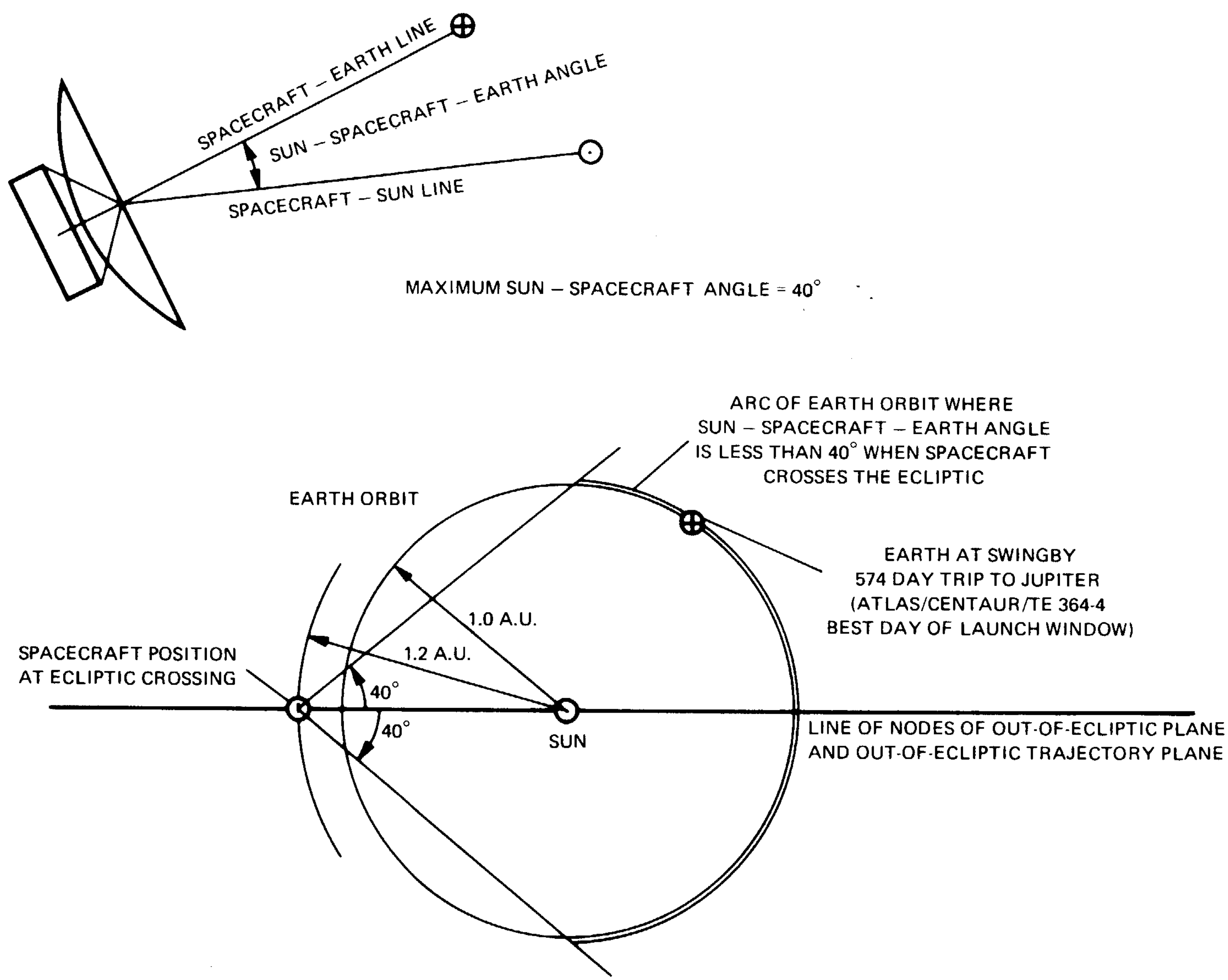

FIGURE 3-26.–SUN–SPACECRAFT–EARTH ANGLE AT ECLIPTIC CROSSING

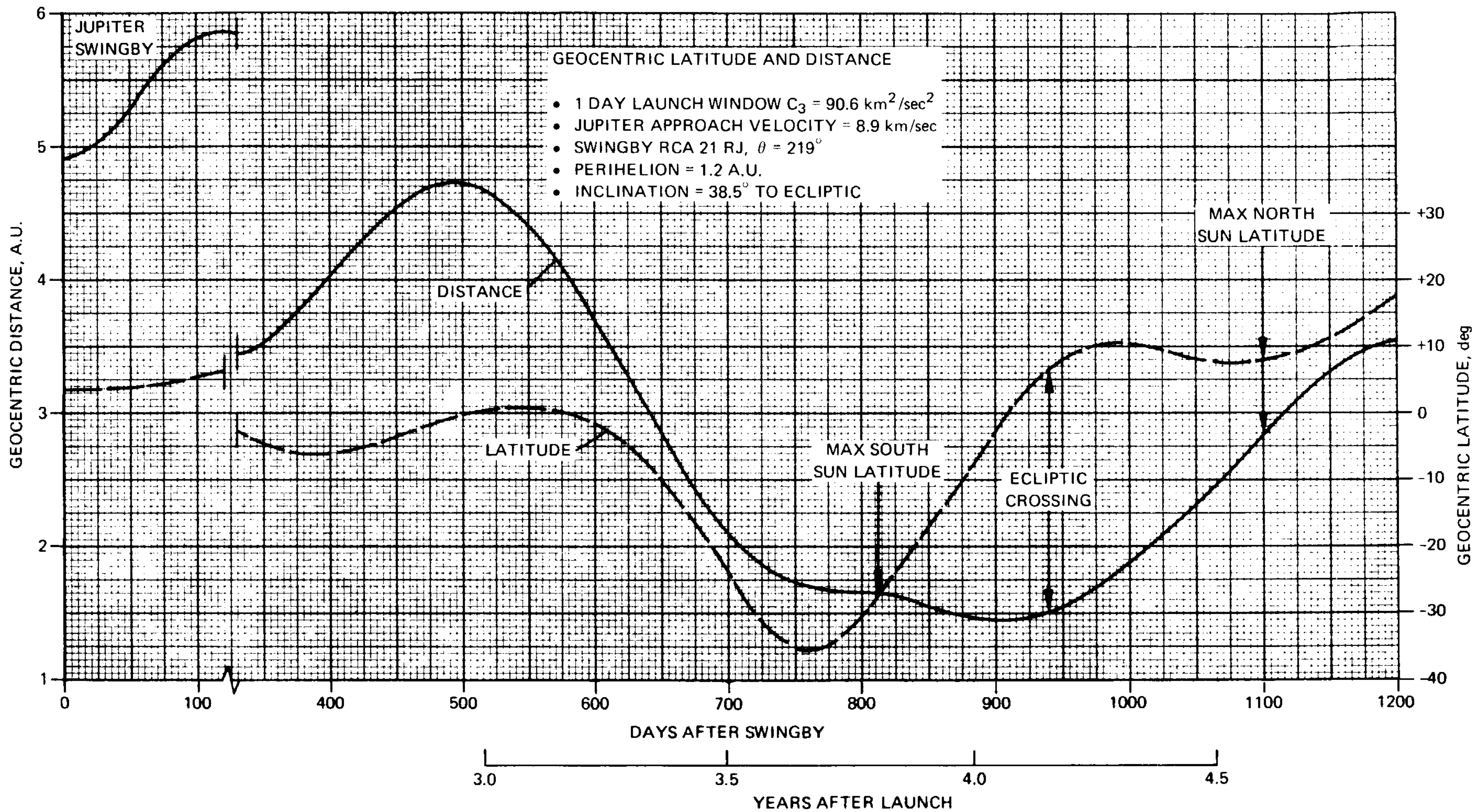

FIGURE 3-27.—NOMINAL OUT-OF-ECLIPTIC TRAJECTORY ATLAS/CENTAUR/TE 364-4

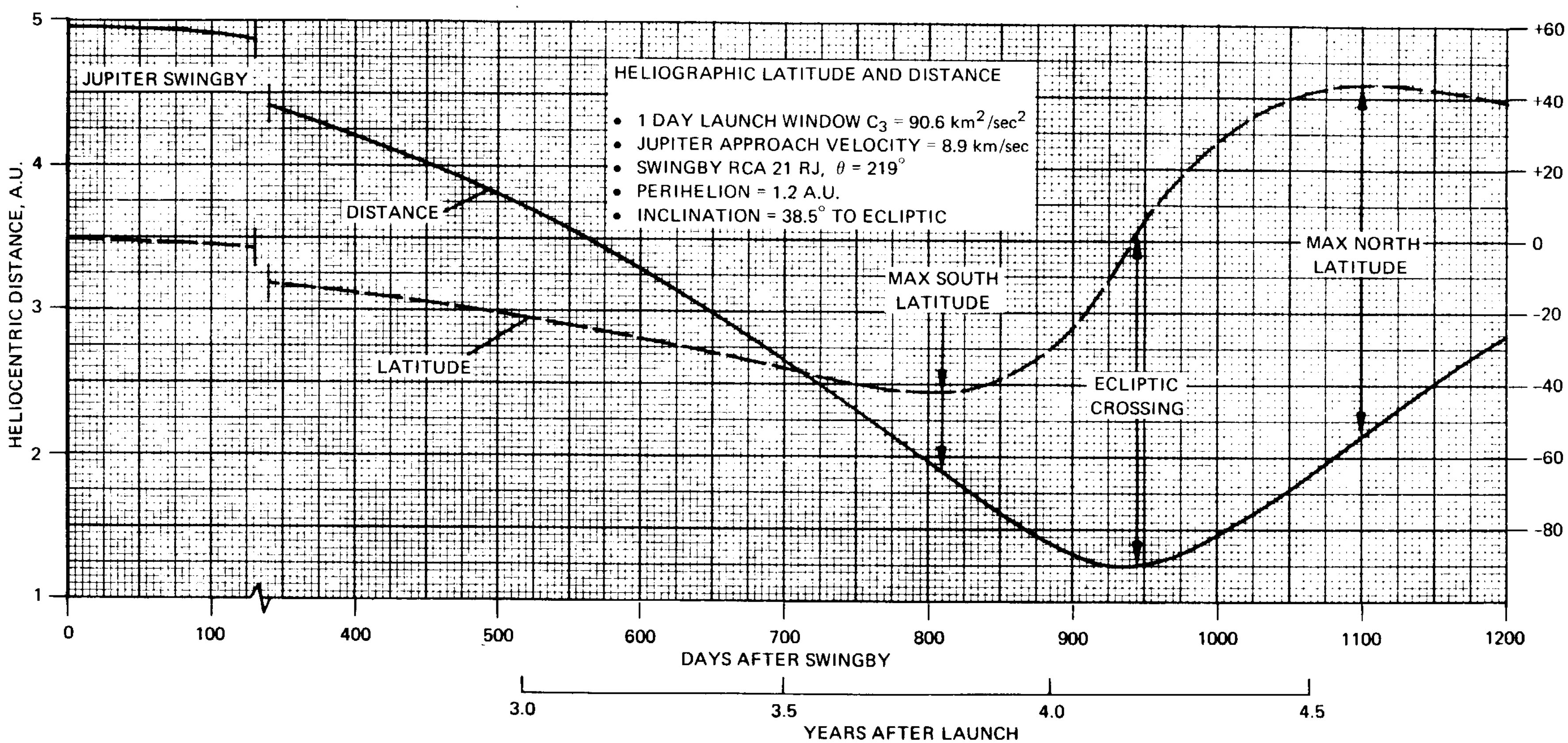

FIGURE 3-28. – NOMINAL OUT-OF-ECLIPTIC TRAJECTORY ATLAS/CENTAUR/TE 364-4

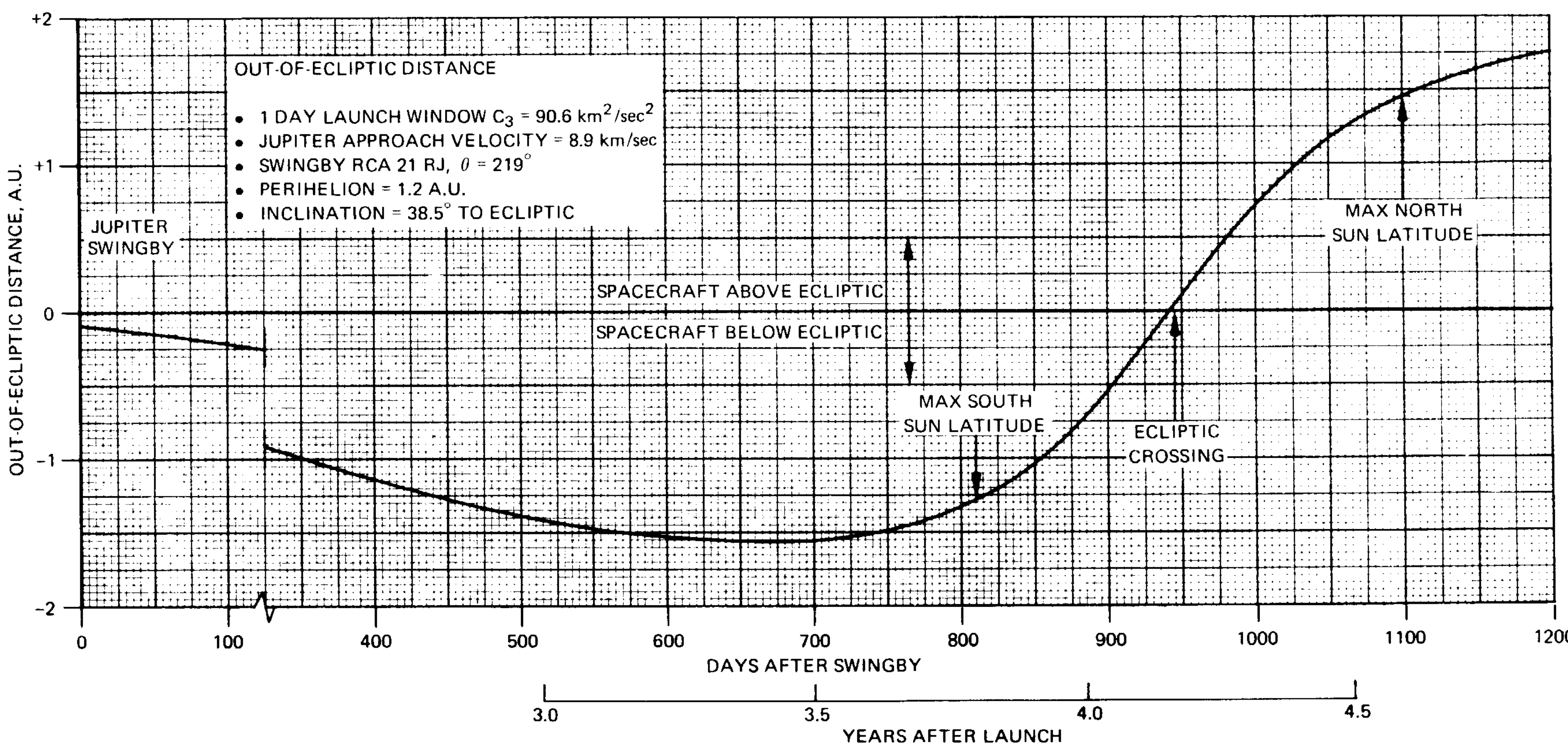

FIGURE 3-29. – NOMINAL OUT-OF-ECLIPTIC TRAJECTORY ATLAS/CENTAUR/TE 364-4

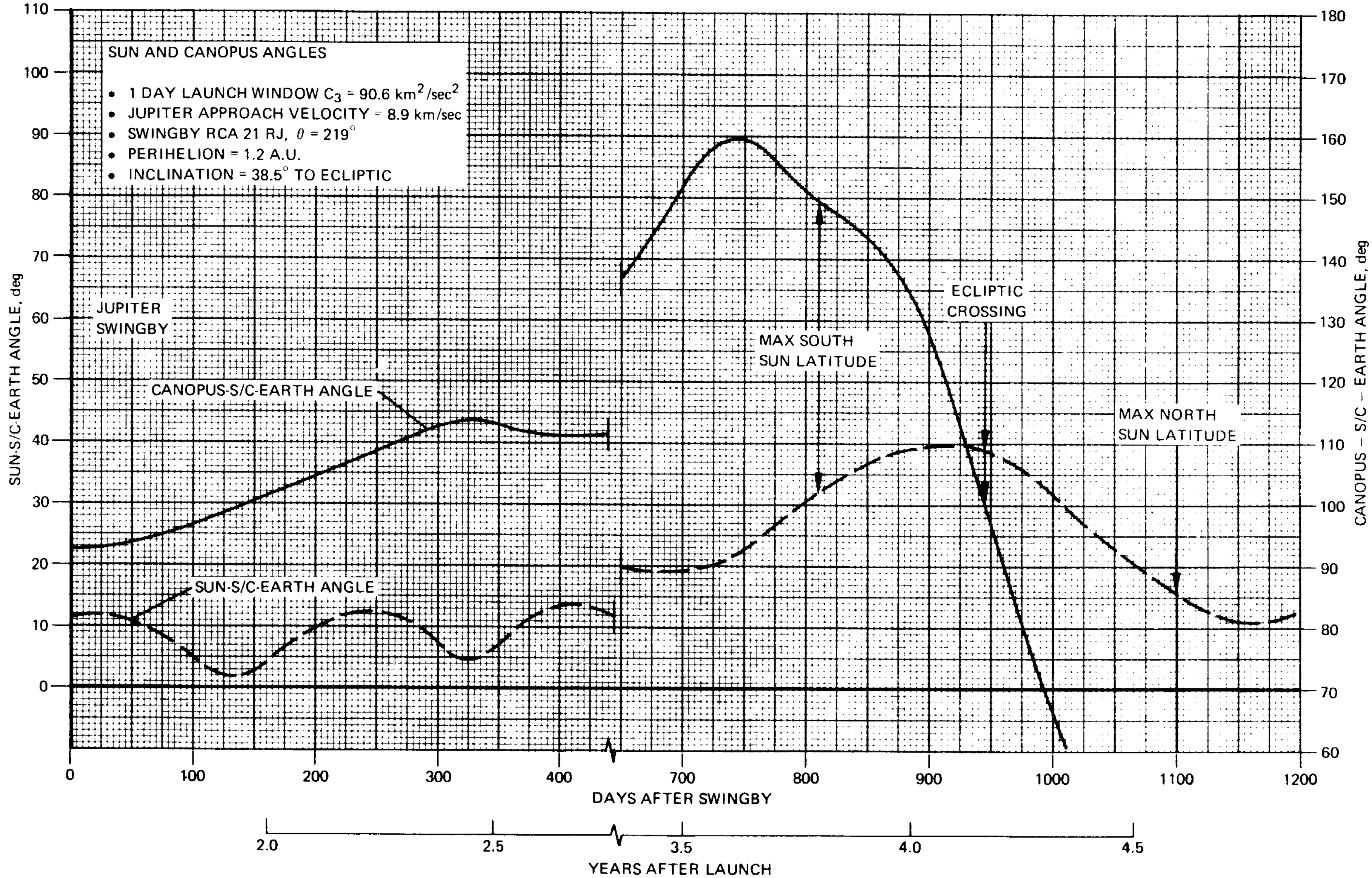

FIGURE 3-30. – NOMINAL OUT-OF-ECLIPTIC TRAJECTORY ATLAS/CENTAUR/TE 364-4

3.7.4 Titan 3C/TE 364-4

3.7.4.1 Launch Energy

The Titan 3C/TE 364-4 is composed of the standard Air Force Titan 3C plus a TE 364-4 upper stage. The estimated launch energy available is plotted in figure 3-21. For the Pioneer spacecraft weight of 555 pounds, the $C_3 = 148.0\ km^2/sec^2$.

3.7.4.2 Launch Window

Figure 3-24 shows the 1974 launch window.

For $C_3 = 148.0\ km^2/sec^2$ the launch window is:

	Date		Trip Time to Jupiter	Jupiter Hyperbolic Approach Velocity
open	JD2442180	13 May	415 days	16.05 km/sec
	JD2442186	19 May		16.65 km/sec
best day	JD2442190	23 May	396 days	16.65 km/sec
	JD2442195	28 May		16.65 km/sec
close	JD2442200	2 June	400 days	16.05 km/sec

The hyperbolic approach velocities at the ends of the window and the best day are from figure 3-8. There is a 9 day window during which the approach velocity is high enough to perform the same mission as the Titan 3D/Centaur/ TE 364-4 launch vehicle.

3.7.4.3 Postswingby Trajectory and Targeting

The Jupiter hyperbolic approach velocities are high enough to allow high postswingby inclinations for reasonable trip times and perihelion distances. For the best 9 day launch window, the approach velocity is high enough to provide an orbit which passes over the Sun poles and has the proper trip time for good Sun-Spacecraft-Earth angles.

Missions flown from this launch vehicle will be similar to missions launched from the Titan 3D/Centaur/ TE 364-4. A summary of trajectory characteristics is:

Launch Day	Swingby RCA	Swingby θ	Postswingby inclination	Postswingby perihelion	Time to ecliptic crossing
13 May	$3.4R_J$	147.9°	86.6°	1.2 AU	2.2 yr
19 May	$3.2R_J$	149.3°	92.5°	1.2 AU	2.2 yr
23 May	$3.2R_J$	149.3°	92.5°	1.2 AU	2.2 yr
28 May	$3.2R_J$	149.3°	92.5°	1.2 AU	2.2 yr
2 June	$3.4R_J$	147.9°	86.6°	1.2 AU	2.2 yr

3.8 ALTERNATE LAUNCH YEARS

3.8.1 1975 Launch Opportunity

The 1975 launch opportunity for Jupiter occurs during June and July. A Titan Centaur launch during this period will conflict with preparations for Viking launches in August and September.

3.8.2 1976 Launch Opportunity

For a launch during the 1976 launch opportunity to Jupiter, the spacecraft will arrive at Jupiter in late 1977. From figure 3-2 the postencounter orbit inclination must be 97.0° for the orbit to pass over the Sun poles.

For Jupiter arrival hyperbolic velocity required is influenced by two factors. At arrival Jupiter is farther from the Sun than it is for a 1974 launch, and thus is moving more slowly. Jupiter's slower speed reduces the required hyperbolic velocity from the 1974 value.

The higher inclination required increases the required hyperbolic arrival velocity. The net effect is that the hyperbolic arrival velocities required at Jupiter swingby are about the same for a 1976 launch as for a 1974 launch.

Figure 3-31 shows the launch date-arrival date curve for a 1976 launch.

Assuming the same spacecraft weight (555 pounds) and Titan 3D/Centaur/TE 364-4 launch vehicle, C_3 = 180 km/sec, and the launch window is:

open	JD 2442980	21 July 1976
close	JD 2443013	23 August 1976
duration	33 days	

The launch window will decrease by one day for each 20 pounds increase in spacecraft weight.

Using launch azimuths from 90° to 110° approximately 3 hours are available during each daily launch period. Parking orbit coast periods will be between 24 minutes and 35 minutes. A period of at least one hour is available each day for launch, during which coast periods will not exceed 30 minutes.

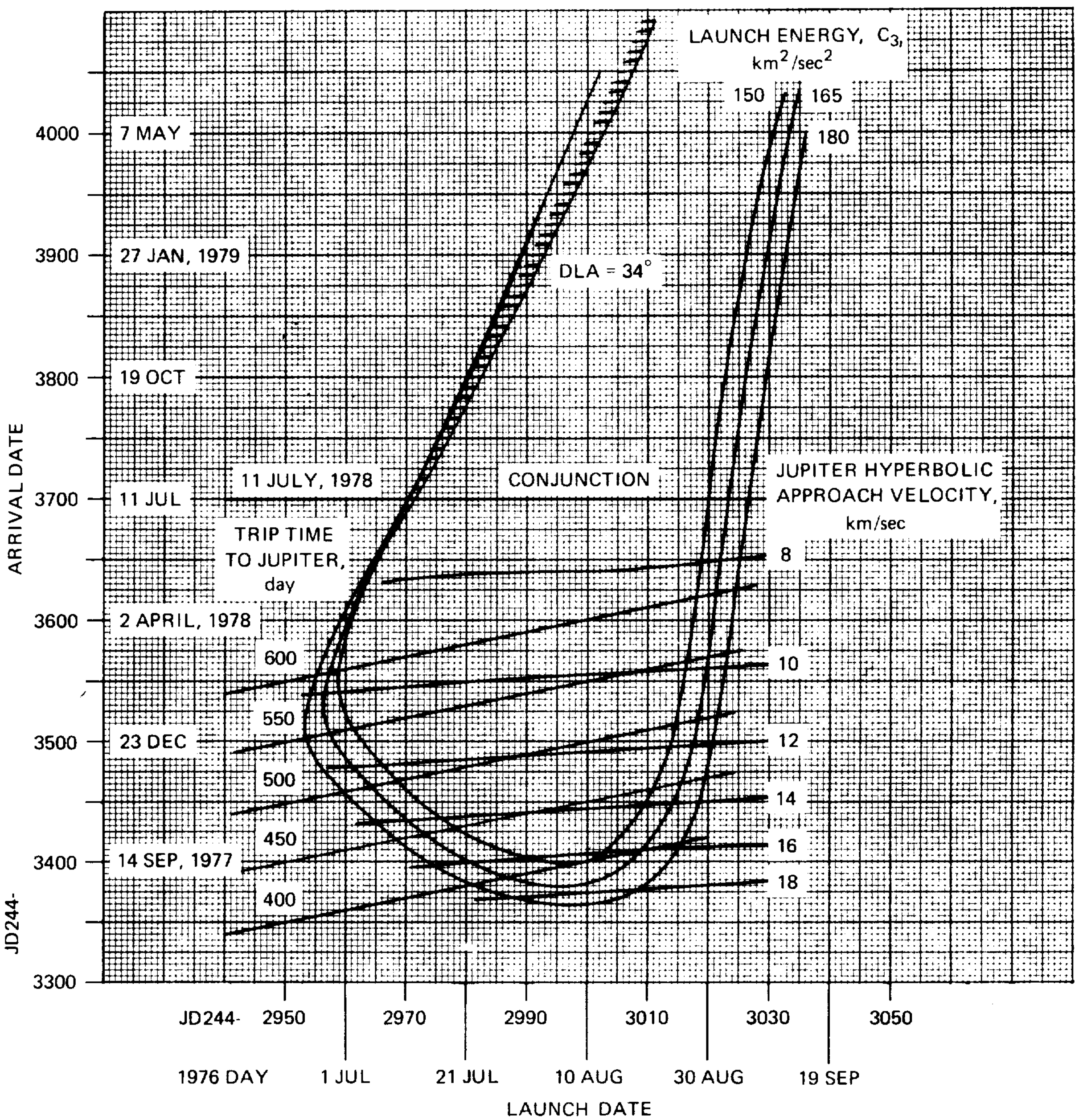

FIGURE 3-31. – JUPITER LAUNCH OPPORTUNITY 1976

Trip time to Jupiter is 400 days for launch at the opening of the window and 379 days for launch at the end of the window.

Targeting at Jupiter will be similar to the 1974 opportunity and the postencounter trajectories to fly over the Sun Pole will be similar but inclined 97° to the ecliptic.

3.8.3 1977 Launch Opportunity

For a launch during the 1977 launch opportunity to Jupiter the spacecraft will arrive at Jupiter in late 1978. From figure 3-2 the postencounter orbit inclination must be 95.5° for the orbit to pass over the Sun poles.

At arrival Jupiter is 5.24 AU from the Sun and moving more slowly than it is for a 1974 launch. Jupiter's slower speed reduces the required hyperbolic arrival velocity from the 1974 value.

The postencounter inclination is slightly greater than that required for a 1974 launch, which slightly increases the required hyperbolic arrival velocity. The net effect is that Jupiter hyperbolic arrival velocities are slightly lower than those required for a 1974 launch.

Figure 3-32 shows the launch date-arrival date curve for a 1977 launch.

Assuming the same spacecraft weight (555 pounds) and Titan 3D/Centaur/TE 364-4 launch vehicle, $C_3 = 180\ km^2/sec^2$, and the launch window is:

open	JD 2443378	23 August 1977
close	JD 2443412	26 Sept 1977
duration	34 days	

The launch window will decrease by one day for each 20 pounds increase in spacecraft weight.

Using launch azimuths from 90° to 110°, approximately 3-1/4 hours are available during each daily launch period. Parking orbit coast periods will be between 29 minutes and 43 minutes.

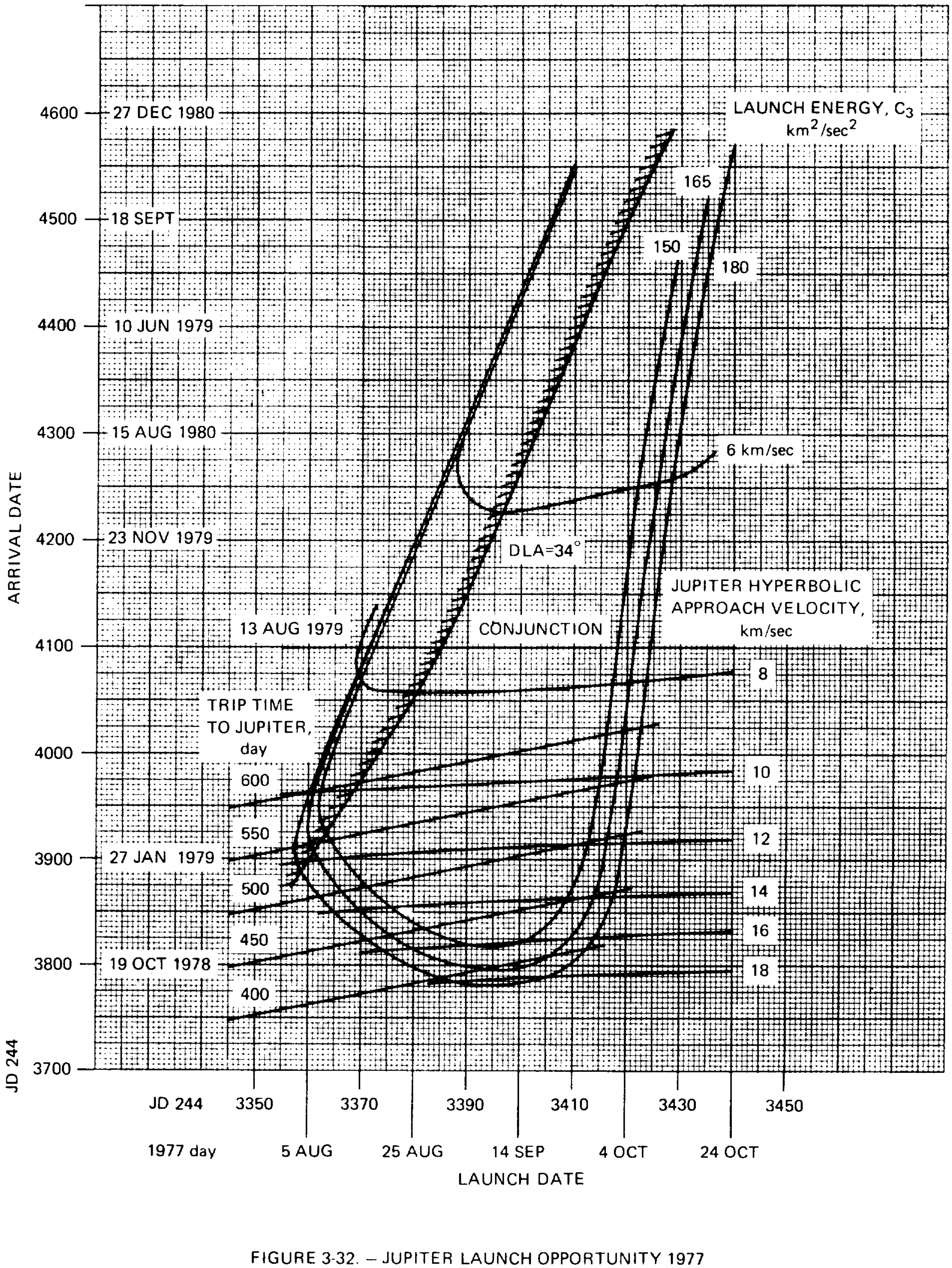

FIGURE 3-32. – JUPITER LAUNCH OPPORTUNITY 1977

Trip time to Jupiter will be 421 days for launch at the opening of the window and 398 days for launch at the end of the window.

Targeting at Jupiter will be similar to the 1974 opportunity with the radius of closest approach slightly larger. The postencounter trajectories to fly over the Sun Pole will be similar but inclined 95.5° to the ecliptic.

4.0 SPACECRAFT ANALYSIS

The spacecraft planned for the Out-of-Ecliptic mission is the Pioneer F/G. This system has been developed for the Jupiter fly-by missions scheduled for launch in 1972 and 1973. The present program includes 3 spacecraft, the F Mission System, the G Mission System, and the Prototype.

This analysis presents a brief description of the F/G design, develops the Out-of-Ecliptic mission requirements as they relate to the spacecraft, and compares these requirements with present systems and subsystem capabilities. As will be seen in the analyses, the requirements of the Out-of-Ecliptic mission compare very closely with those now defined for the F and G missions.

The feasibility of using the F/G Prototype spacecraft for this mission is evaluated, and plans for the refurbishments required are presented.

Reliability estimates are given based on an extrapolation of the present F/G predictions.

Potential spacecraft design changes aimed at increased mission lifetime or improved system reliability are discussed.

4.1 PIONEER F/G SYSTEM DESCRIPTION

The design configuration of the Pioneer F/G spacecraft is illustrated by figure 4-1.

The spacecraft at launch weighs 555 pounds, which includes approximately 60 pounds of scientific instruments.

The spacecraft is spin-stabilized in flight, with the spin axis parallel to the axis of the high gain antenna, which is directed at Earth in the normal cruise mode. Precession and spin rate control are provided by small hydrazine thrusters, which also serve midcourse propulsive maneuver requirements.

The spacecraft equipment compartment is located directly beneath the high gain antenna reflector. It contains the electronic units of the spacecraft subsystems, scientific instruments, and the propellant tank. Mechanical louvers located in the compartment floor provide thermal control.

Power is provided by four SNAP-19 RTG's (Radioisotope Thermoelectric Generators). The RTG's are mounted on telescopic booms which are deployed radially shortly after powered flight is concluded.

Communications to and from the spacecraft utilize S-band carrier frequencies with PCM/FSK/PM uplink modulation and PCM/PSK/PM downlink modulation. Downlink information rates are selectable over the range 16-2048 bps.

The spacecraft carries three communication antennas. The high-gain antenna is a 9-foot parabolic reflector which is mounted forward with its axis parallel to the spacecraft spin axis. A forward tripod structure supports the high-gain antenna feed, as well as a medium-gain horn antenna with its beam center canted from the spin axis to provide scanning as the spacecraft rotates. The third antenna, an omni-directional log-conical spiral is mast-mounted on the aft end of the spacecraft.

Spacecraft attitude control is achieved through error signals derived from a conical scan of the up-link r-f beam. In this manner, antenna attitude (and spacecraft attitude) are determined, and appropriate thrusters are operated to precess the spacecraft to a proper alignment with the beam. Roll orientation is provided by solar and stellar sensors.

A system block diagram for the F/G spacecraft is shown in figure 4-2.

Further material describing the Pioneer spacecraft and the F and G Missions may be found in the Pioneer Program Document P-201.

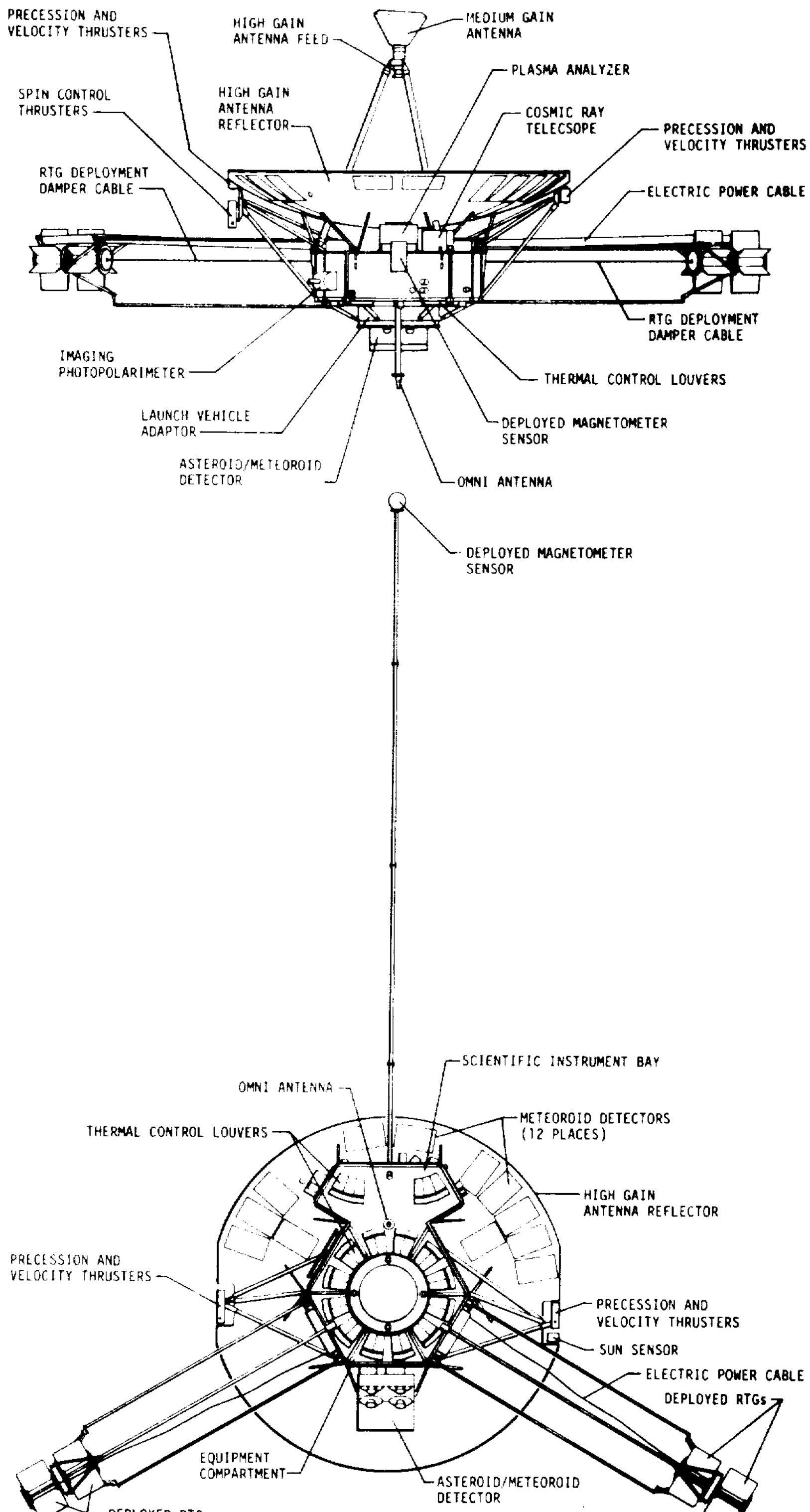

Figure 4-1. Pioneer F/G Spacecraft

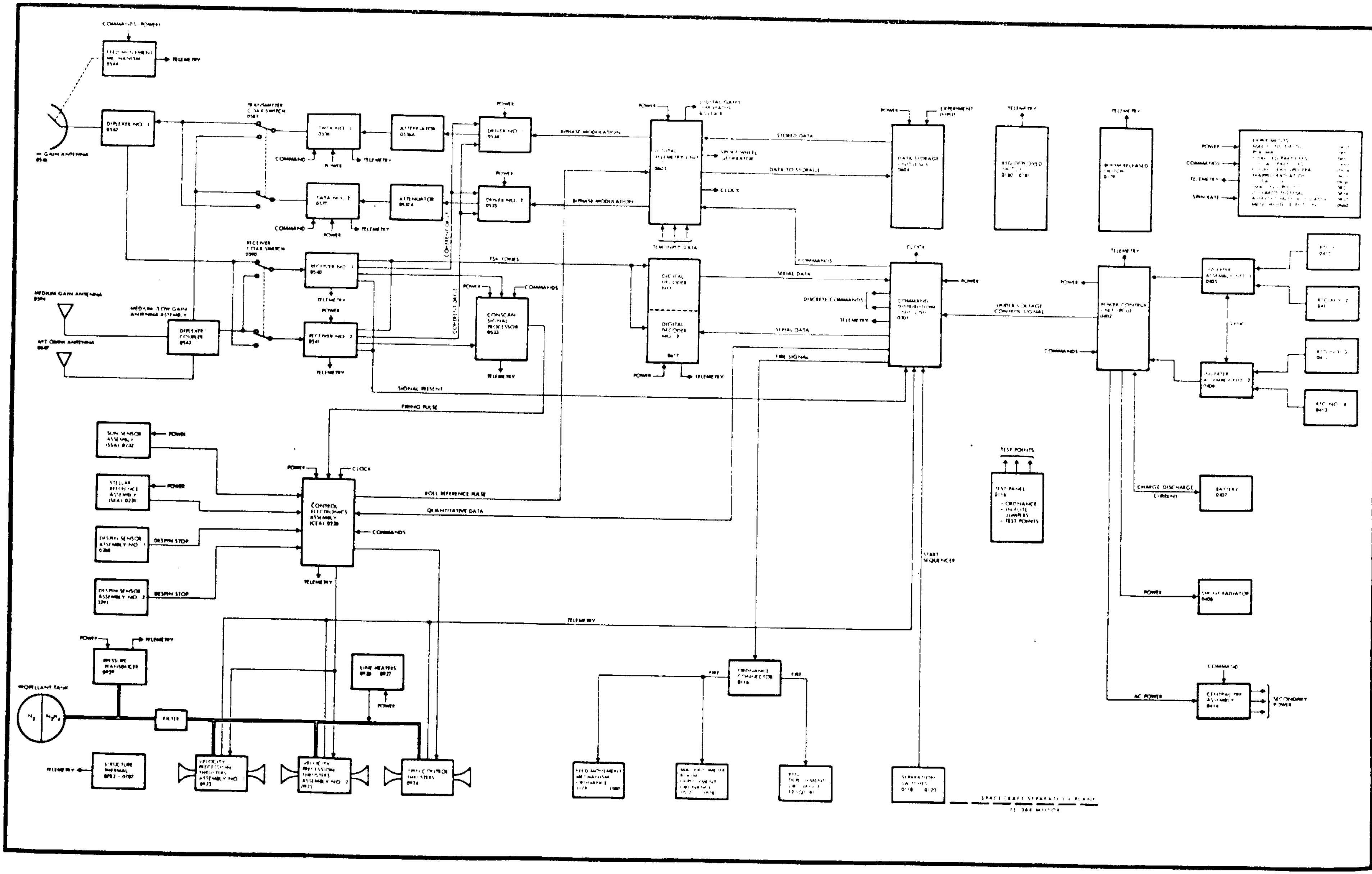

Figure 4-2.- Pioneer F/G Spacecraft System Block Diagram

4.2 SYSTEM REQUIREMENTS FOR THE OUT-OF-ECLIPTIC MISSION

4.2.1 System Environments

4.2.1.1 Launch Dynamics

The Pioneer F/G hard environment specifications have been defined to include the levels anticipated for either the Atlas/Centaur/TE-364-4 or Titan/Centaur/TE-364-4 launch vehicles. Spacecraft system qualification testing is being performed at the levels for vibration and acoustic noise shown in Table IV-1.

4.2.1.2 Asteroids and Meteoroids

The Pioneer F mission has been analyzed for possible damage to the spacecraft due to collision with interplanetary particules. Results have shown probability of survival greater than 0.96, based on collision velocities up to 25 Km/Sec. This analysis would be directly applicable to an Atlas/Centaur launched Out-of-Ecliptic mission. For a Titan/Centaur launch, the trip time through the Asteroid belt is shorter and collision velocities slightly higher. At 2.8 AU from the Sun (approximate midpoint of the belt) an Atlas launched spacecraft will have a velocity of 19.4 Km/Sec. At this same location for a Titan launched mission, spacecraft velocity will be nearly 24.5 Km/Sec. The relative velocity of the Titan launched spacecraft with respect to particles within the belt is 18 Km/Sec, which is still well below the collision velocities considered in the penetration and damage analysis.

4.2.1.3 Jupiter Radiation Belts

The Jupiter swingby distance for an Atlas launched Out-of-Ecliptic mission is on the order of 20 R_J. At this radius, the potential for radiation damage is negligible. Titan trajectories require closer passage to the planet, at distances which radiation models show significant flux.

Total electron and proton fluence for Titan trajectories with Jupiter passages at the 3.2 and 5.4 R_J distances were calculated using the radiation model by Devine (JPL). The results along with fluence estimates are summarized and compared with expected F mission fluence in Table IV-2.

The results show nominal electron and proton fluence for the Titan 3.2 R_j trajectory to be less than that predicted for the Pioneer F trajectory by a factor of 3. This is owed principally to the faster passage of the planet; also the higher inclination and slightly greater swingby distance. The nominal radiation fluence calculated for a Titan 5.4 R_j trajectory are further reduced; approximately 2 orders of magnitude down from the Pioneer F trajectory for electrons and 1 order of magnitude for protons.

These comparisons serve to show the range of fluence which could be expected with varying trajectory parameters. To interpret fluence in terms of damage potential to the spacecraft requires an analysis of potentially vulnerable devices and their particular application.

TRW has investigated electron radiation damage to the MOSFET (Metal-oxide-silicon field effects transistor) which is considered one of the more susceptible devices. In laboratory tests MOSFETS were subjected to a 1.2×10^{12} e/cm^2 at 1.0 Mev equivalent exposure. Resistance was measured and interpreted in terms of required performance. Safety factors were calculated for a range of applications, which showed for a worst case condition, a value of +2.1. Safety factors for more typical applications were determined to be +7.2. Other data have been compiled which suggest these devices will sustain partial damage at fluences of approximately 10^{10}e/cm^2, and serious permanent damage at levels 10^{11} and above.

Proton radiation models for Jupiter are not established with the degree of certainty which is acknowledged for the predictions for electron radiation. The model used for this analysis (by Dr. Devine, JPL) was selected because of its relevance to other work, at ARC.

Susceptibility of electronics to proton radiation has been estimated for a number of devices. Some types of transistors begin to show effects beginning with a fluence of approximately 10^8 p/cm^2, and may sustain serious permanent damage at levels above 10^{10} p/cm^2.

A Germanium low frequency transistor used in the spacecraft inverter has an estimated damage threshold at 5×10^8 p/cm^2. In this application, the device is used as a switch so that minor degradation of the transistor would normally not affect the circuit function.

Shielding for high energy protons was not incorporated with the present F/G design. No shielding allowance or consideration has been introduced in this analysis.

TABLE IV-1. PIONEER QUALIFICATION LEVEL ENVIRONMENTS

a. Random Vibration

Vibration Axis	Test Duration Min. Each Axis	Frequency Hz	PSK Level g^2/Hz
All Three Axes	4	20-100	0.056 at 100 Hz with roll-off of 6 dB per octave from 100 to 20 Hz
		100-1000	0.056
		1000-2000	0.056 at 1000 Hz with roll-off of 12 dB per octave from 1000 to 2000 Hz

b. Sinusoidal Vibration

Vibration Axis	Frequency Hz	Acceleration g's (0-peak)
Thrust	5-12 12-50 50-200	3.0 3.0 2.25
Both Lateral	5-10 10-22 22-200	1.95 1.95 1.50

c. Acoustic Noise

Octave Band Center Frequency (Hz)	Sound Pressure Level (dB) (ref. 0.0002 microbar)
16	122
31.5	128
63	135
125	140
250	144
500	142
1000	137
2000	132
4000	132
8000	132
Overall:	148

Duration: 2.0 Minutes

TABLE IV-2.- ELECTRON AND PROTON FLUENCE FOR JUPITER RADIATION BELT PASSAGE

	PIONEER F	OUT-OF-ECLIPTIC TRAJECTORY TITAN LAUNCHED	OUT-OF-ECLIPTIC TRAJECTORY TITAN LAUNCHED	OUT-OF-ECLIPTIC TRAJECTORY ATLAS LAUNCHED
Trajectory				
Swingby Distance*	3.0 R_j	3.2 R_j	5.4 R_j	20 R_j
Target Latitude**	14.2°	30.7°	35.5°	41°
Electron Fluence (3.0 Mev Equiv.)				
Nominal	3×10^{10}	7.4×10^{9}	2.5×10^{8}	$<10^{6}$
Upper Limit	1.3×10^{12}	6.4×10^{12}	3.0×10^{11}	$<10^{9}$
Proton Fluence (20 Mev Equiv.				
Nominal	7.5×10^{9}	2.7×10^{9}	6.5×10^{8}	$<10^{7}$
Upper Limit	4.2×10^{12}	2.6×10^{12}	1.6×10^{12}	$<10^{10}$

*Measurement from planet center

**With respect to Jupiter Equatorial plane

Though the proton radiation may indeed be as high as the model predicts, it must be remembered that the basis for the model is very insecure. Data which can be used to establish actual proton radiation levels will not be available until the Pioneer F flyby which will nominally occur 6 months before a 1974 Out-of-Ecliptic launch opportunity. At that time, and even up to the time of executing final midcourse maneuvers, a decision could be made to target for a greater swingby radius to avoid the deeper radiation belts.

4.2.2 Performance Requirements and Constraints

4.2.2.1 Mission Duration

The Out-of-Ecliptic Mission duration can vary from approximately 3 to nearly 5 years, depending on the launch vehicle energy, choice of Jupiter swingby conditions, and a definition of the point in the trajectory where scientific objectives will have been met. For purposes of this study, that point has been taken to be the second pole crossing. The trip times associated with trajectories considered in this study are shown in Table IV-3.

4.2.2.2 Power

The power requirements for the Out-of-Ecliptic Mission have been taken as the values now budgeted and specified for Pioneer F and G. A breakdown of the power budget is shown by Table IV-4.

The nominal power required for normal mode operations is budgeted at 100.2 watts. The peak normal mode requirement, which takes momentary surges and commanding loads into account is 112.3 watts. In addition to the normal mode requirements, there are three commandable operation modes which have raw power requirements as follows:

Conscan Coarse Tracking - 16.1* watts

Conscan Fine Tracking - 22.6* watts

Propulsive ΔV Maneuvers - 9.9 watts

Power requirements vs. capabilities are discussed in detail in Section 4.3.1, Power Subsystem.

4.2.2.3 Thermal Control

The primary thermal requirement is that the spacecraft equipment compartment located beneath the high gain antenna be maintained between 0° and 90°F throughout

* Includes operation of antenna feed bias mechanism.

TABLE IV-3.- TRIP TIMES FOR OUT-OF-ECLIPTIC TRAJECTORIES

Trip Times (Years)

Trajectory	Earth to Jupiter	Jupiter to First Sun Pole Crossing	First Sun Pole to Second Sun Pole	Total Trip Time
Titan - 3.2 R_j	1.09	1.86	.81	3.76
Titan - 5.4 R_j	1.09	3.73	.77	4.59
Atlas - 20 R_j	1.78	2.08*	.74*	4.60

*Times referred to maximum solar latitude since Sun pole crossing is not achieved with these trajectories.

TABLE IV-4. SPACECRAFT POWER BUDGET (WATTS)

	SPACECRAFT		EXPERIMENTS		TOTAL	
	Nominal	Peak	Nominal	Peak	Nominal	Peak
Power Control Unit						
Loads	31.8	33.9	25.2	27.7		
Cable Loss	.3	.4	.3	.3		
PCU Electronics & Shunt Reg.	3.1	3.1	0	0		
D.C. Bus Input	35.2	37.4	25.5	28.0		
Rectifier-Filter Loss	2.5	2.8	1.8	2.0		
	37.7	40.2	27.3	30.0	65.0	70.2
Central Trans. Rectifier						
Loads	12.7	15.2				
Losses	6.5	7.9				
	19.2	23.1			19.2	23.1
AC Required	56.9	63.3	27.3	30.0		
Inverter Loss	9.1	10.1	4.4	4.8		
Cable Loss	1.7	2.8	.8	1.3		
Raw Power Rqmts.	67.7	76.2	32.5	36.1	100.2	112.3

the mission. The spacecraft temperature control capability is sensitive to Sun angle, particularly at solar distances near or less than 1 A.U.

A constraint of 1.2 AU has been imposed in trajectory analysis to limit minimum perihelion distance and to provide for Earth-spacecraft phasing which maintains a solar aspect that avoids the side Sun condition. This and other aspects of thermal control are discussed in Section 4.3.2, Thermal Subsystem.

4.2.2.4 Propulsion (Spacecraft)

The onboard propulsive energy requirements for a Pioneer Out-of-Ecliptic flight derive from three types of spacecraft maneuvers:

a. Midcourse velocity changes

b. Attitude orientation maneuvers

c. Despin and spin rate control

The velocity change requirement is the most significant of the three. An analysis of the errors associated with the Pioneer F Mission launch vehicle guidance and the resulting injection velocity errors shows a requirement for a maximum 100 meters/second velocity correction capability to achieve target conditions for a proper swingby at Jupiter. Estimates for targeting accuracy capability are given by an error ellipsoid with a semi-major axis of approximately 2400 km. This analysis would apply directly to an Atlas launched Out-of-Ecliptic trajectory.

Injection velocity errors for Titan trajectory can be conservatively estimated from the Pioneer F data by a ratio of the respective V_∞ values. Calculations based on this show a maximum velocity change requirement for a Titan launched trajectory of 140 meters/second. Consideration for the planetary quarantine bias is included.

There is no requirement anticipated for ΔV maneuvers after the Jupiter swingby.

Precession angle change requirements for an Out-of-Ecliptic mission are estimated as follows:

Initial reorientation	-	180°
Earth tracking - outbound trajectory	-	120°
Midcourse maneuvers (2)	-	360°
Earth tracking - inbound trajectory	-	120°
Earth tracking - solar swingby	-	270°
		1050°
Contingency	-	450°
		1500°

The despin requirement, based on a maximum anticipated rate change is 60 rpm.

Propulsion requirements are summarized as follows:

			Titan	Atlas
a.	Midcourse	-	140 mps	100 mps
b.	Precession	-	1500 deg.	1500 deg.
c.	Despin	-	60 rpm	60 rpm
d.	Spin Control		30 rpm	30 rpm

The capability of the Pioneer spacecraft to meet these requirements is discussed in Section 4.3.4, Propulsion Subsystem.

4.2.2.5 Communications

Communication and data rate requirements are based on F/G spacecraft operational needs and the F/G science payload. Data rate requirements range from a maximum at Jupiter encounter to reduced levels during interplanetary cruise. Spacecraft control can be accomplished satisfactorily at the minimum bit rate of 16 BPS.

The maximum communication range for the Out-of-Ecliptic trajectories considered will occur during the Jupiter encounter. Since encounter is also the occasion for maximum science bit rate, the communication requirements are established by this condition.

The Pioneer F/G spacecraft has a data rate capability of 1024 BPS at Jupiter. This assumes the use of the DSN 210' antenna systems.

The uplink communication requirements for this mission are well within spacecraft and DSN capability.

Further description of the Pioneer communications subsystem and its capabilities is given in Section 4.3.7.

4.2.3 System Constraints

4.2.3.1 Spacecraft Gross Weight

The F/G spacecraft total lift-off weight is now estimated to be 555 lbs. A breakout of the major elements is given in Table IV-5. For a Titan launch, payload weight is not a limiting factor. The Titan can readily achieve the required velocities with payloads greater than 600 lbs, and still preserve a launch window on the order of 20 days. Atlas trajectories, however, show reduced ex-ecliptic inclination with increased payload, to the ratio of approximately 1 degree per 7 lbs.

TABLE IV-5. PIONEER F/G WEIGHTS

Subsystem/System	Weight (lb.)
Electrical Power (less RTG's)	38.3
Communications	22.6
Antennas	45.0
Data Handling	11.8
Electrical Distribution	35.3
Attitude Control	12.1
Propulsion (Wet)	83.2
Thermal Control	13.8
Structure	102.7
Balance Weight	6.2
Ballast for c.g. Adjust.	2.6
Spacecraft System	373.6
RTG System (GFE)	117.6
Scientific Instruments	63.9
Gross Spacecraft (Total)	555.1

4.2.3.2 Dynamical Mass Properties

The mass distribution of the Pioneer F/G spacecraft provides for an inertia ratio (ratio of spin moment of inertia to the average transverse moment of inertia) of about 1.87 in cruise mode (after appendage deployment). This allows the spacecraft to maintain attitude within an allowable drift rate of 0.2°/day maximum at a spin rate of approximately 5 rpm. Equipment placement is arranged to locate the radial center of gravity near the spacecraft geometric centerline, which is the desired spin axis but more importantly, the axial c.g. must be in the plane of the appendages to minimize wobble induced during deployment. Appendage deployment radius, mass, and angular positions were selected for null products of inertia.

4.2.3.3 Magnetics

In consideration of the scientific payload, principally the magnetometer, magnetic property specifications have been levied on the Pioneer spacecraft. The component of the magnetic field in the spin axis direction at the scientific magnetometer sensor that is induced by the spacecraft, excluding that of the scientific instruments shall not exceed:

a. 0.03 γ stray field due to the energized spacecraft.

b. 0.04 γ due to remanence after demagnetization.

c. 0.25 γ due to remanence after an exposure of no less than 25 gauss along each axis.

4.3 SUBSYSTEM ANALYSIS

This section contains the analyses and assessments of the present Pioneer design capabilities in relation to the requirements of an Out-of-Ecliptic mission as developed in section 4.2.

4.3.1 Power Subsystem

Two elements of the power subsystem were found to be limiting for the requirements of an Out-of-Ecliptic Mission. They are RTG performance and battery lifetime.

4.3.1.1 RTG Performance

The F/G spacecraft uses four SNAP 19 RTGs as the basic system power source. These RTGs nominally provide an output of 39 watts each at the beginning of life. They would normally be activated 9-10 months prior to launch, so that their output at the beginning of the mission would be approximiately 37 watts each, for a total raw power capability of 148 watts. Figure 4-3 shows the mission power requirements with estimated SNAP-19 performance.

The F/G system design was based on an RTG performance degradation of approximately 0.2 watt per month-per unit, so that at the end of a 2 year mission period, the raw output from each RTG would be reduced to 32.2 watts, or for four units, 128.8 watts. If the 0.2 watt per month-per unit rate is extrapolated out to a 5 year mission period, performance then would be at 25 watts per unit or 100 watts total. This level is still sufficient to power the spacecraft and experiments in normal operating modes, assuming that peak pulse loads are carried by the battery as the design intends.

A more conservative prediction for RTG performance has been developed based on data obtained from tests of electrically heated generators by the contractor, Isotopes Inc. This data shows a slightly higher rate of degradation, such that the power available from four units would be down to 100 watts after about 3.8 years instead of 5 years as predicted from the earlier estimates.

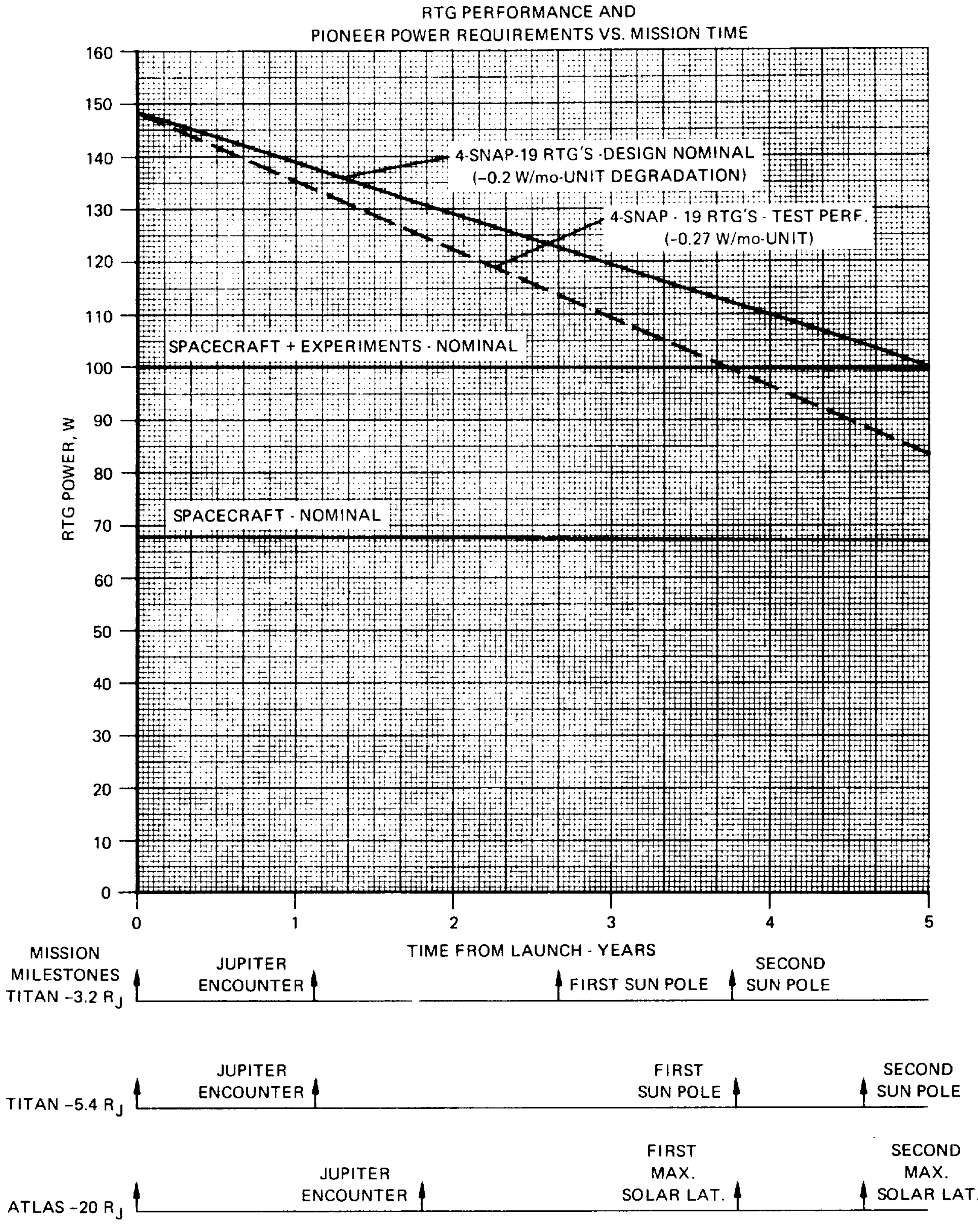

FIGURE 4-3. – PIONEER POWER REQUIREMENTS VS. RTG PERFORMANCE

Other more pessimistic predictions can be derived by factoring in data from recently delivered RTG flight units. This however is an anomolous condition which is now being investigated by the AEC and should not be the basis for normal performance predictions.

Using the degradation rate of 0.27 watts per month-per unit, it is seen that a mission duration of between 3.5-4 years is the limit at which nominal requirements would begin to exceed RTG power available. Operations beyond this point would require that power be allocated on a time share basis. This mode could be used to extend useful mission time by an additional 1-2 years.

The Out-of-Ecliptic missions under consideration would have maximum durations of 3-5 years, depending on the choice of launch vehicle, Jupiter swingby conditions, and a definition of the useful mission end point. The trajectory associated with the Titan IIID/Centaur/TE-364-4 launch vehicle and 3.2 R_J swingby distance at Jupiter would pass over the Sun and around to the second polar crossing 3.7 years after launch. For this mission, operations would not be substantially affected by power limitations.

Engineering improvements to the present RTG design which would reduce the rate of power degradation were examined. This was in addition to the malperformance studies now being conducted by the AEC. The results of this effort are given in Section 4.7.

4.3.1.2 Battery

The F/G spacecraft uses a silver-cadmium battery. Batteries of this design have excellent magnetic qualities, but are limited in lifetime. Analysis based on the experience of other missions indicates only limited probability of specified performance after 2.5 years.

The consequence of an in-flight battery failure as the subsystem is presently designed, would be an inability to handle short term loads which exceed the RTG output. This would result in an automatic shut-down of power to certain spacecraft elements, which could later be restored by command up to the limits of RTG output.

Alternates to the present battery design were examined, including:

a) Improvements to AgCd battery lifetime through more sophisticated temperature control.

b) Replacement of the AgCd battery with a NiCd unit.

c) Fly the present AgCd battery, but provide for an in-flight activation to be performed later in the mission when RTG power has degraded and the battery is required for peak loads.

The detailed results of this investigation are also presented in section 4.7.

4.3.2 Thermal Subsystem

The spacecraft thermal subsystem consists of insulation materials, insulative coatings, and an array of mechanical louvers located in the floor of the equipment compartment. The subsystem is designed to maintain temperatures within the equipment compartment between 0° and 90°F for the conditions of the F and G missions. Out-of-Ecliptic trajectory features which warrant examination because of circumstances differing from the F and G trajectories are:

a) Longer mission duration.

b) Trajectory return toward the Sun.

c) Spacecraft attitude with respect to the Sun.

A preliminary investigation of thermal coatings used on the Pioneer F/G spacecraft has indicated that the white low-absorptivity coating used on the inside of the antenna dish will degrade as a result of the long exposure to ultraviolet radiation. Near the end of the mission this might cause higher temperatures than otherwise expected, particularly with a direct Sun-on attitude.

This question was investigated by assuming that the absorption coefficient would increase by a factor of two due to coating degradation. The antenna dish was assumed to be an adiabatic plate in a Sun-on attitude at 1 AU. The temperature of the inside surface was calculated to be 153°F, which compares with a temperature of approximately 35°F for non-degraded coatings as measured in test. The temperature of 153°F is conservative since in flight some heat will be transferred to the cooler surface at the back of the antenna. The effect of this temperature increase is not considered significant in terms of either structural softening or system heat balance.

Spacecraft attitude with respect to the Sun is a more critical constraint. The spacecraft is somewhat marginal for maintaining equipment compartment temperature within prescribed limits with a side-Sun attitude and solar distances inside 1 AU. The spacecraft is normally oriented with the high gain antenna facing Earth. For the Pioneer F and G missions, the spacecraft spin axis-Sun angles, except for brief times early in the mission and during midcourse maneuvers are less than 30°. This allows the spacecraft equipment compartment to be continuously shadowed by the dish, and therefore not subjected to the side-Sun condition. The configuration of the spacecraft affords reasonable antenna shading for Earth-Spacecraft-Sun angles up to approximately 45°. Out-of-Ecliptic trajectories have been selected for Earth-Spacecraft phasing which maintains a forward Sun aspect, within 40° of the spin axis through the second Sun pole crossing. In addition, a somewhat arbitrary constraint on minimum Perihelion distance of 1.2 AU has been used which takes surface coating degradation into account and allows some side-Sun attitude at the minimum solar distances should Earth-Sun phasing be unfavorable.

The 1.2 AU minimum perihelion constraint is consistent with the science objectives. Closer perihelion distances reduce the trip time from north Sun pole to south Sun pole and thereby increase the rate of solar latitude change with time. The solar latitude change rate associated with perihelion distances greater than 1 AU permits adequate differentiation of spatial vs. temporal phenomena.

A separate analysis was made of the temperatures which would be expected for selected spacecraft elements at solar distances closer than 1 AU, assuming a forward Sun aspect (equipment compartment shaded). It was concluded that a solar distance of 0.8 AU represents the limiting condition. At this distance, certain spacecraft components would be operating at or near their maximum allowable temperatures. The calculations for this condition indicate a temperature for the high gain antenna of approximately 240°F. Also, the Sun facing side of the RTG's would be expected to approach 375° at the fin root, which is near the critical temperature of the connector seal.

4.3.3 Structure Subsystem

The spacecraft structure is designed to withstand environments which include the launch dynamics of the Titan IIID/Centaur/TE-364-4 configuration as well as the Atlas/Centaur/TE-364-4.

The critical element of the present design is the RTG boom and boom support structure. Test results and structural analysis indicate the present RTG weight is within 10% of the allowable limits for the RTG support structure. This is discussed further in connection with possible RTG modifications in Section 4.7.

Structural limitations for loads within the equipment compartment are not restrictive. The spacecraft was designed to carry a 90-pound instrument payload. The F/G instrument complement weighs approximately 60 pounds.

4.3.4 Propulsion Subsystem

The mission propulsion requirements as developed in section 4.2.2 are based on total midcourse velocity increments of 140 meters per second (Titan) and 100 meters per second (Atlas), total precession angle change of 1500 degrees (Titan and Atlas), despin of 60 rpm (Titan and Atlas) and spin control of 30 rpm (Titan and Atlas).

4.3.4.1 Velocity Change

The Pioneer F/G propulsion system uses hydrazine monopropellent thrusters for velocity change and attitude/spin control. Velocity changes are normally accomplished by an on-axis maneuver, wherein the spacecraft spin axis is aligned with the velocity vector to be added, and the thrusters are operated continuously for the total ΔV required. An alternate technique is also available, termed an off-axis maneuver, which would normally be reserved for relatively small velocity adjustments at communication distances beyond those afforded by the omni-low gain antenna. In this maneuver, thrusters along and perpendicular to the spin axis are pulsed at the appropriate phase in the spacecraft spin cycle, so as to produce, over a series of spin cycles, the two in-plane components of the desired velocity vector. Precession induced by the unbalanced moments is compensated by opposing thrusters. Thus the spacecraft attitude is maintained in relation to the Earth for high gain antenna operations, and the velocity increment is applied at some desired angle in relation to that attitude.

4.3.4.2 Despin/Spin Control

One despin/spin control thruster cluster assembly is mounted on the spacecraft near the outer edge of the high gain antenna. After injection, a despin maneuver of approximately 60 rpm is accomplished with one steady-state burn. After deployment of the RTG's and the magnetometer, the spacecraft despin/spin control is used to maintain spacecraft spin rates of approximately 5 rpm. Spin control is maintained by pulsing; however, if a major change in spin rate is required, a steady state burn may be used.

4.3.4.3 Precession

The spacecraft has two thruster cluster assemblies mounted on a platform approximately 50 inches from the spin axis. The thrusters are arranged such that precession around the spin axis may be accomplished in either direction since each assembly has two thruster nozzles 180° apart.

4.3.4.4 Characteristic Performance

Isp performance characteristics for the various maneuvers are given below:

			Isp
a)	On-axis ΔV - continuous thruster firing	-	215 sec.
b)	Off-axis ΔV - pulsed thruster firing	-	$\sim$70 (approx. equivalent)
c)	Precession	-	140 sec.
d)	Despin	-	215 sec.
e)	Spin Control	-	140 sec.

The Isp given for off-axis maneuver has been adjusted to account for the three major penalties involved; pulsed operation which results in a less efficient expansion through the nozzle, velocity increment by components where the sum of the components is necessarily greater than the resultant, and additional propellant required for precession compensation.

4.3.4.5 Propellant Requirements

Nominal propellant requirements for the total mission were calculated for both the Atlas launched and Titan launched trajectories:

			Titan	Atlas
a)	Despin (60 rpm)	-	.7 lb.	.7 lb.
b)	On-axis ΔV	-	33.8 lb. (140 mps)	23.9 lb. (100 mps)
c)	Precession (1500 deg.)	-	9.8 lb.	9.8 lb.
d)	Spin Control (30 rpm)	-	2.3 lb.	2.3 lb.
	Total		47.6 lbs.	36.7 lbs.

The quantity of propellant carried is nominally 60 pounds, with approximately 2 pounds budgeted for leakage and ullage, making 58 pounds available for the mission.

C-2

4.3.5 Attitude Control Subsystem

The spacecraft attitude control subsystem is designed to perform the following maneuvers:

a) Despin

b) Spin Control

c) Conscan

d) ΔV (on-axis)

e) ΔV (off-axis)

The despin function provides a reduction in spin rate after injection from the nominal 60 rpm required during 3rd stage burn to approximately 22 rpm, where deployment of the RTG's and magnetometer can occur. The spin control function is performed in response to ground command to achieve an increase or decrease in spin rate as required. The Conscan function gives closed loop precession pointing of the spacecraft spin axis toward Earth, using signals derived from uplink beam scan. The attitude control subsystem controls the direction and duration of ΔV maneuvers for both the on and off-axis techniques. The subsystem also provides roll reference signals for orientation of the scientific instruments.

The subsystem performance capabilities match or exceed Out-of-Ecliptic mission requirements as the hardware now exists, however certain functional redundancies which are available for the F and G missions are marginal or unavailable in certain instances with Out-of-Ecliptic trajectories. In particular, the low angle (0-12°) Sun sensor channel of the Sun sensor assembly, is nonredundant. However, functional redundancy in this sector is achieved through use of the Stellar Reference Assembly. For the F and G trajectories, the spacecraft spin axis is always in the ecliptic plane, effectively normal to the reference star, Canopus. As the spacecraft approaches Jupiter, and Earth-Spacecraft-Sun angles become small, the principal reference would become the star sensor. With an Out-of-Ecliptic trajectory, attitude conditions prior to the Jupiter encounter are much the same as for F and G. However, as the spacecraft returns toward the Sun, the ex-ecliptic distance, while maintaining Earth pointing, changes spacecraft attitude in relation to Canopus and ultimately the star passes out of sensor limits (90° ±19° from spin axis). For the

Titan 3.2 R_J trajectory, the Earth-Spacecraft-Sun angle is always greater than 15° at times when Canopus is not visible so that a reference redundancy is still available through the Sun sensor itself. (See figure 3-10.) In the case of the Titan 5.4 R_J and Atlas trajectories (figures 3-20 and 3-30) the Earth-Spacecraft-Sun angle goes below 7° for some periods during which Canopus is not in view. In those instances, roll reference is dependent on the single Sun sensor element.

Two potential solutions have been examined. One is a hardware change which provides a duplicate 0-12° sensor channel in the Sun sensor assembly, thus giving complete redundancy for all Earth-Spacecraft-Sun angles. This is a fairly simple change, would involve approximately 0.3 pounds of additional hardware and an additional 0.25 watts of power and would affect only the Sun sensor assembly package. An alternative to hardware change would be to attempt to use stars in addition to Canopus for roll reference. However, the SRA (Stellar Reference Assembly) threshold is set so that stars with less than 50% of Canopus' silicon band intensity are excluded. Also, the spectral content and intensity of Canopus were used as the basis for selection of the SRA detectors. Successful detection and lock on other stars is therefore a more difficult operational problem. This is particularly true if star acquisition is attempted without benefit of solar reference. This could be the condition of required use, if the star reference is thought of as a backup to the Sun sensor, and its function initiated at the occurrence of a Sun sensor failure.

The greater accuracy of the star sensor for roll reference in this sector (approximately ±0.5° vs. ±1.25° for Sun sensor) is not critically important for this mission.

4.3.6 Data Handling Subsystem

The capabilities of the data handling subsystem are completely adequate for the requirements of an Out-of-Ecliptic mission. Details of the interface between this subsystem and the science instruments are presented in Section 4.5.

4.3.7 Communications Subsystem

The spacecraft communications subsystem performs the following functions:

a) Receives and demodulates commands modulated in a PCM/FSK/PM format from the Deep Space Stations (DSS).

b) Modulates and transmits to the DSS scientific and engineering data in a PCM/PSK/PM format.

c) Generates, demodulates and processes a conical scan error signal on the RF uplink carrier that is utilized by the attitude control subsystem to precess the spacecraft spin axis toward Earth in a closed loop attitude control mode.

d) Radiates a noncoherent RF signal with no uplink signal present to permit acquisition of the spacecraft by the DSS.

e) Provides a phase coherent retransmission after acquisition of an uplink signal such that two-way doppler measurements can be made at the DSS.

Major subsystem parameters are summarized in Table IV-6.

The communication subsystem down link bit rate capabilities are given by Figure 4-4. (Reference Earth spacecraft distance plot, Figures 3-13, 3-17, 3-27.) For the maximum range case, which is during Jupiter encounter, the maximum data rate is 1024 bits per second. This is based on use of the spacecraft high gain antenna and the DSN 210' antennas.

The communication range for this mission decreases after Jupiter swingby. This has led to the consideration of reduced TWT power as a means of alleviating system power requirements, particularly later in the mission. At present, the two TWT's are both rated at 8-watts output. Maximum bit rates could still be achieved inside 3 AU with a 4-watt TWT. A discussion of the factors involved in this alternate is presented in Section 4.7.

The potential input power savings of a 4-watt vs. an 8-watt TWT in terms of RTG power is approximately 20 watts.

TABLE IV-6. MAJOR COMMUNICATION SUBSYSTEM PARAMETERS

Uplink Frequencies	
Receiver 1	2110.584105 MHz
Receiver 2	2110.925154 MHz
Downlink Frequencies	2292.037037 MHz 2292.407407 MHz ($\frac{240}{221}$ of uplink f)
Effective Radiated Power (ERP) on Spin Axis	
High Gain Antenna	70 dBm
Medium Gain Antenna	47 dBm
Low Gain	35 dBm
Bit Rates	16 through 2048 bps (powers of 2)
Uplink Antenna Beamwidths (3 dB)	
High Gain	3.5°
Medium Gain	32°
Low Gain	≈120°
Downlink Antenna Beamwidths (3 dB)	
High Gain	3.3°
Medium Gain	29°
Low Gain	≈120°
Antenna Pattern Alignment (Relative to +Z Axis)	
High Gain (Normal Mode)	0°
High Gain (Conscan Mode)	1°
Medium Gain	9.3°
Low Gain	180°
Pointing Accuracy (Conscan)	
High Gain Antenna	0.3° (90%
Medium Gain Antenna	1.3° confidence)

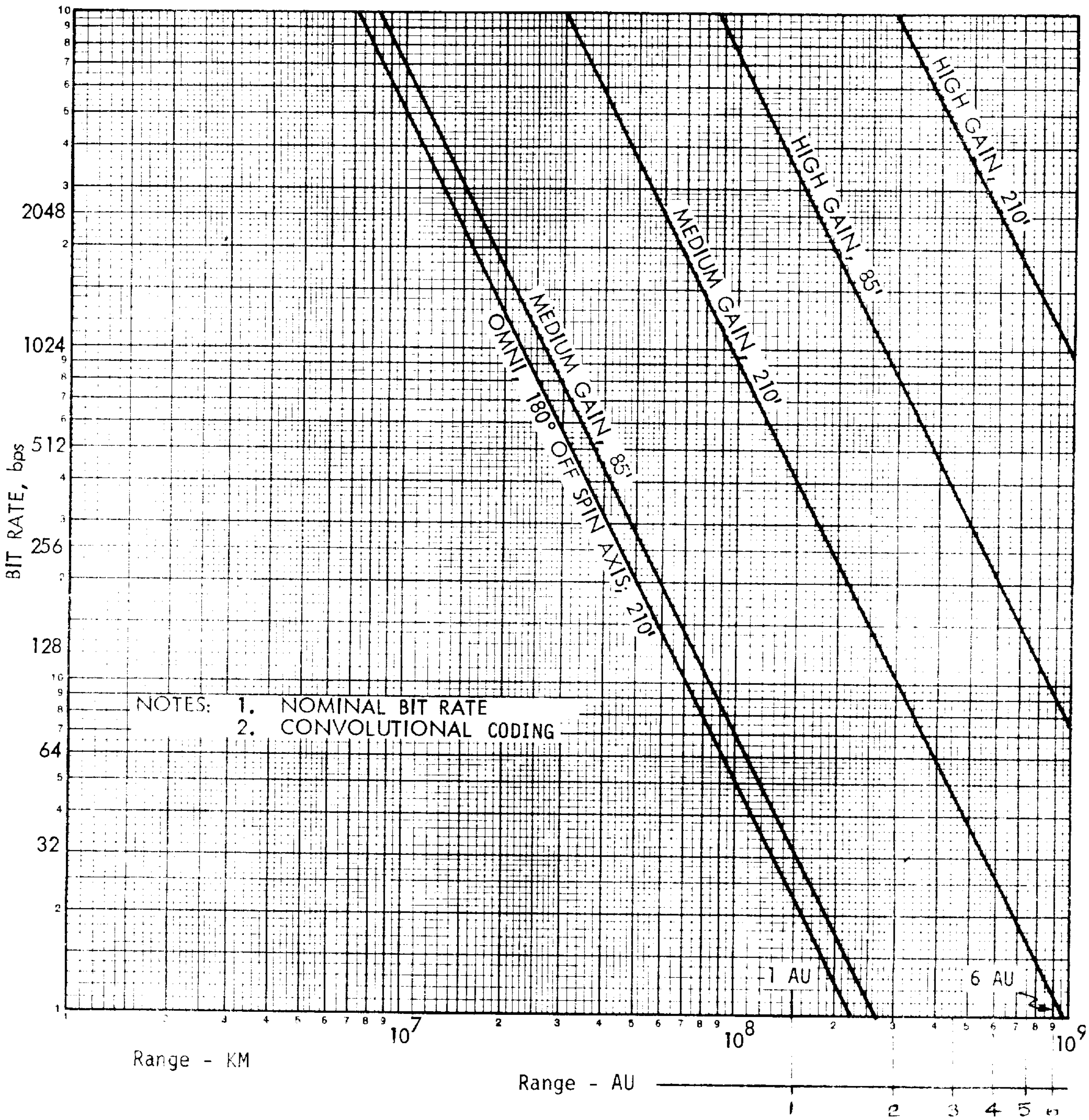

Figure 4-4.- Pioneer F/G Telemetry Bit Rate Vs. Communication Range

4.4 SCIENCE INSTRUMENT INTEGRATION

The science payload instruments as they are now carried on Pioneer F and G are shown in Figure 4-5. Features of the instrument-spacecraft interface are described below.

4.4.1 Mechanical Interface

Instruments can be mounted both internal and external to the equipment compartment, however, the mounting distribution and arrangement must satisfy system mass and c-g constraints. The F/G configuration has achieved a mass distribution such that the c-g is near the axial center line, in the RTG-Magnetometer Boom plane (within close tolerances). Changes to the instrument payload would require appropriate balance consideration; either compensating balance weights, or a redistribution of the existing units within and on the compartment.

4.4.2 Electrical Interface

Science payload power is distributed to each instrument by an individual fused branch circuit. Voltage supplied is 28 VDC, regulated to $\pm$0.5 percent short term with an allowable $\pm$1.0 percent drift.

4.4.3 Thermal Interface

Temperatures in the vicinity of scientific instruments that are mounted within the equipment compartment or to the exterior surfaces of the compartment are maintained between 0°F and 90°F when the instrument is powered. Instruments not mounted within the equipment compartment or to the exterior surfaces (within the insulation blanket) of the compartment must provide their own temperature control.

4.4.4 Data Handling and Control

The spacecraft accepts from the scientific instruments information in digital, analog, or state form, converts the analog and state information to digital form, and arranges all information in an appropriate format for time multiplexed transmission to Earth or storage onboard the spacecraft. The spacecraft also supplies the instruments with various timing and spacecraft operational status signals as well as functional commands. A telemetry

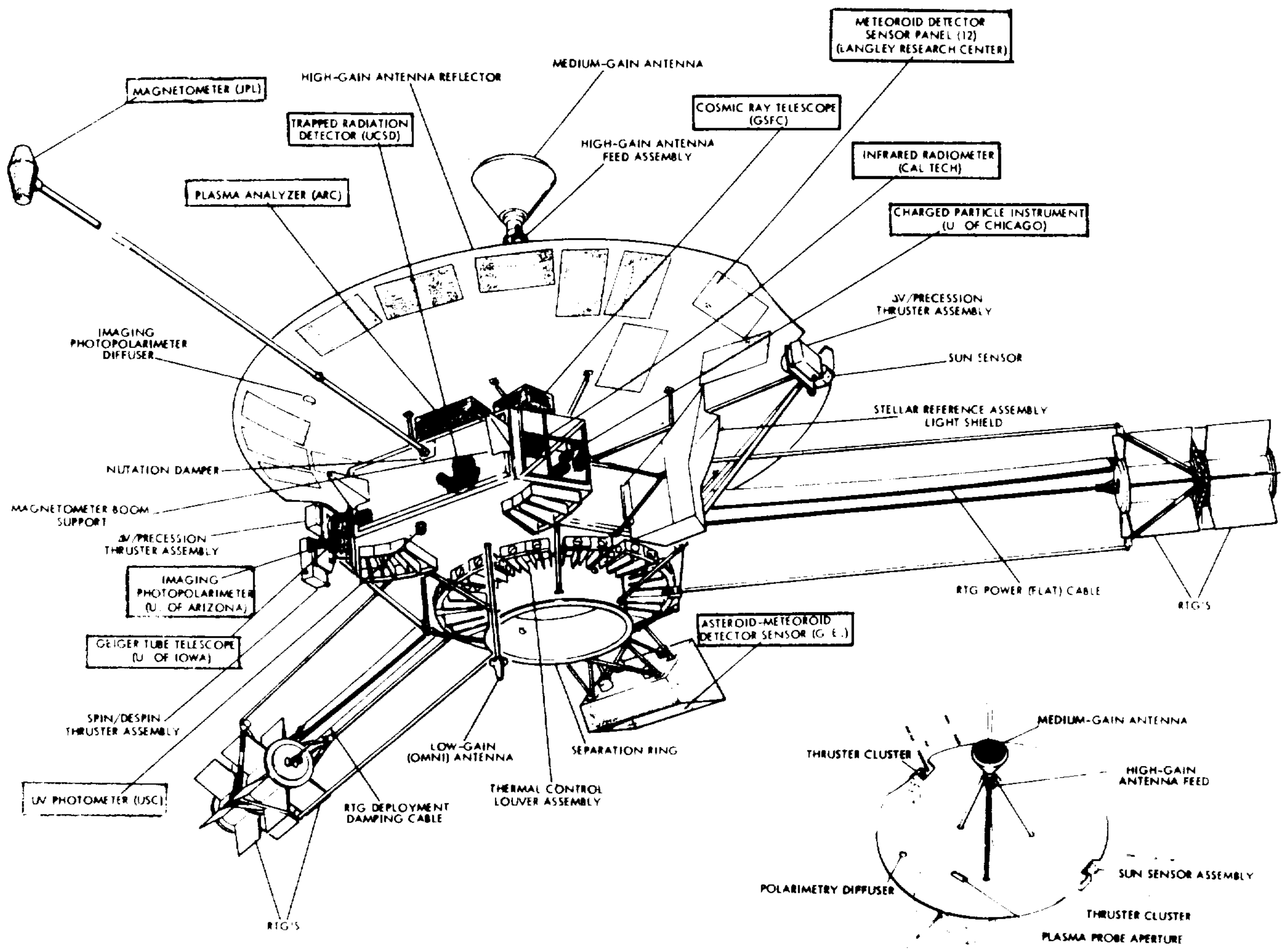

Figure 4-5. Pioneer F/G Spacecraft - Experiment Instrument Location

word in the main science formats (Formats A, B, and D) will consist of 3 binary bits. A telemetry word in the engineering formats (Formats C-1, C-2, C-3, and C-4) and the subcommutated science formats (Formats E-1 and E-2) will consist of 6 binary bits.

The data subsystem assembles the information from the instruments into frames composed of a series of 192 bits. The data subsystem supplies timing and other signal information to the instruments, including:

a. Frame, subframe, word rate, and bit shift pulses.
b. Clock pulse at 32.768 kH .
c. Clock pulse at 2048 H .
d. Bit rate status of DTU.
e. Data format selected.
f. Roll index pulse.
g. Spin sector pulse (up to 512 sectors/revolution).

Fifty individual function commands for the instruments have been assigned for the F/G payload. The spacecraft has the capability for 70 such commands. The spacecraft Data Handling Subsystem has capacity for science words in addition to the present F/G instrument payload. Spare channels now exist in both the A and B science formats and several of the D formats are available for assignment.

These additions would require harness modifications, but would not effect the electronic units which are presently designed with patch panels and modular plug-in capability.

4.5 SYSTEM RELIABILITY

4.5.1 Reliability Predictions

Reliability figures for the F and G missions have been calculated which show a system reliability prediction of 0.7933, based on a 900-day mission. The system number derives from predictions of subsystem reliability and other system elements as follows:

Antennas	0.9906
Communications	0.9724
Attitude Control	0.9838
Data Handling	0.9419
Electrical Power	0.9894
Electrical Distribution	0.9678
Structural	0.9931
Thermal	0.9994
Propulsion	0.9724
Meteoroid Analysis	0.9612

These reliability predictions are based on the successful operation of all major units as required to perform any of the mission critical or mission significant functions. The unit reliability assessments were in turn based on generic part failure rates adjusted as a function of the applied thermal and electrical stress ratios which relate to flight conditions. An estimate for the probability of mission success as a function of mission time for the Out-of-Ecliptic flight is shown by the graph of Figure 4-6. Here the F/G reliability data is plotted at the 900-day point, and an extrapolation is made to extend the graph as a function of the expression $e^{-\lambda t}$ where t is now 5 years. This assumes Out-of-Ecliptic Mission environments and operating conditions to be comparable to the F/G situations with only an extension to mission duration. This extrapolation shows a system reliability after 5 years of 0.62. Reliability considerations unique to the Out-of-Ecliptic mission are discussed in the following paragraphs.

4.5.2 Jovian Radiation Belts

Passage of the radiation belts near Jupiter was not addressed in the F/G reliability studies since post Jupiter spacecraft survival is secondary to primary mission objectives at the Planet. These objectives include requirements for deep measurements in the radiation belts. Hence an F or G mission could sustain spacecraft radiation damage at Jupiter and still accomplish desired objectives. The Out-of-Ecliptic mission, on the other hand, absolutely requires successful transit of the belts with the spacecraft and the science instruments emerging fully intact.

An accurate prediction of mission reliability as a function of the radiation damage potential is a complex problem requiring data which define the circuit effects of the damage degraded devices. This was not undertaken on a system wide basis, however, an estimate was made of the power subsystem reliability effect due to damage to the PNP Germanium transistor discussed in Section 4.2.1.3. The proton fluence estimated for the Titan 3.2 Rj mission based on the Devine model was 2.7×10^9 p/cm^2 and 6.5×10^8 p/cm^2 for the Titan 5.4 Rj mission. If the damage suffered by this exposure were to double the potential failure rate for this device, then power subsystem reliability prediction would reduce from 0.76 to 0.51 for a 5-year mission life. The assumptions for this calculation are completely arbritrary, however, it does illustrate reliability prediction sensitivity to degrading environmental conditions.

Atlas launched missions would pass the planet at distances ($\sim$ 20 Rj) which are for practical purposes, beyond the radiation hazard and consequently encounter should have no effect on system reliability.

4.5.3 Life Limited Items

The Pioneer spacecraft design has used parts selection criteria which impose long life, low failure rate requirements, the results of which are evidenced by the success of Pioneers VI, VII, VIII and IX. These criteria generally exclude parts which would present any wear-out or life limited problems within a 5-year mission duration. Potential exceptions in the Pioneer F/G design were identified and are discussed in sections below.

4.5.3.1 Battery

The life limits of the AgCd battery have been presented in Section 4.3. Of importance from a system reliability standpoint is that a major failure of the battery, would not directly affect the spacecraft operation except when peak power loads exceed RTG output. At these times, with the battery inoperative, it would be necessary to time-share the available RTG power. This is an operating contingency which the spacecraft is capable of conveniently responding to since on-off commands are provided for nearly all important power consuming functions. Options for an improved battery design are discussed in Section 4.7.

4.5.3.2 Propellant Tank Diaphragm

There was an initial concern that the rubber diaphragm inside the propellant tank may lose its properties with long term exposure to hydrazine. A considerable amount of compatibility testing of diaphragm materials has been conducted by JPL and by Pressure Systems Incorporated.

The rubber material which has been found to be the most suitable is Ethylene Propylene (EPT-10) which is now used in the Pioneer tank. A requirement for the use of EPT-10 is that the operating temperature should not exceed 100°F. Lower temperatures are more favorable. Except for brief periods early in the trajectory, the highest temperatures expected for the Out-of-Ecliptic Mission, assuming the side Sun constraint is observed, would be less than 100°F. Pressure Systems Incorporated who supply the tank and the diaphragm for the Pioneer F/G spacecraft have been conducting tests with EPT-10 and hydrazine for over two years. EPT-10 samples which have been tested for this period of time show no effects (interaction with the hydrazine or loss of properties). Pressure Systems Incorporated has expressed a complete confidence in the suitability of the material for a 5-year mission period. This test data, along with a recognition of the favorable temperature environment afforded by the Out-of-Ecliptic trajectory suggests that this is an unlikely failure possibility.

4.5.4 System Redundancy

It is noted that the F/G design provides complete design or functional redundancy in each of the following critical areas:

a. Communications
b. Command and Data Handling
c. Propulsion Electronics, Thrusters
d. Attitude Sensors

4.6 PROTOTYPE REFURBISHMENT

The principal questions regarding use of the F/G Prototype spacecraft for the Pioneer H Mission to which the study investigations have been addressed are the following:

a. What is the present status of the spacecraft relative to a fully flight worthy condition?
b. Is a fully flight worthy condition achievable?
c. What would be the basis for and elements of refurbishment criteria for the spacecraft.
d. Is refurbishment of the Prototype cost effective?
e. What effect does this use of the prototype have on the F and G missions?

Answers to these and other questions have been developed and are presented in the subsequent paragraphs of this section.

4.6.1 Present Status

The spacecraft is currently undergoing the F/G system qualification tests. The configuration includes certain exceptions to a flight worthy status, primarily units which have been designated for test only because of unscreened parts or similar deficiencies. These items are identified in the program as ALU's (Accepted for Limited Use Only). In addition to these known flight status exceptions, considerations have been expanded to include those elements of the spacecraft which may suffer reduced function or capacity as a result of the qualification test experience. This has led to an examination of the test program for an assessment of potential effects which is given by Table IV-7.

4.6.2 Refurbishment Criteria

In the establishment of criteria for refurbishment, the primary concern is that the reliability of the spacecraft is not reduced due to age sensitive components or test and fabrication history. The intent of the criteria presented is to ensure that the prototype spacecraft meets the level of reliability established for the Pioneer F/G Spacecraft.

TABLE IV-7.

ASSESSMENT OF THE EFFECTS OF SYSTEM QUALIFICATION TESTING OR THE PROTOTYPE SPACECRAFT

TEST	CRITICAL LEVELS	POTENTIAL EFFECTS	ASSESSMENT
a. Thermal-Vac	0.9 AU Solar Intensity 10^{-5} mm Hg 17 days	Over-heat of Electronic Components	·Level is not considered critical Should not cause damage or degradation.
b. EMC	Injected noise on signal lines	Damage to front end diodes-etc	·Test levels are not severe. Design philosophy provides for conservative de-rated device application.
c. Vibration	3 g's p-p 5-50 Hz (Thrust axis) 0.056 g^2/Hz @100 Hz (3 axis) Grad/sec^2 20-60 Hz Torsional	Structural Damage	·Random levels are not severe ·Sine levels may generate damaging responses in low frequency elements of the structure. RTG and Mag. booms are likely high load items. ·Torsional test loads may overstress the RTG booms.
d. Acoustic Noise	148 db overall 16-8000 Hz 2 min	Structural Damage	·Thin structures may be affected.

Considering technical adequacy commensurate with successful Pioneer F/G operation as the primary constraint, criteria used for refurbishment and/or replacement of prototype spacecraft components are based on coordinated evaluations of:

a. Fabrication and environmental test history.
b. Engineering orders not incorporated into the prototype units.
c. Components authorized for limited usage in prototype units, which require replacement.
d. Components and materials which are sensitive to shelf life and require replacement and/or special storage consideration.
e. The level of retest and period of retest required for each of the spacecraft components.

Table IV-8 has been prepared to show the results of applying these criteria to the prototype hardware for the major elements of each spacecraft subsystem and represents the basic spacecraft refurbishment plan. In nearly all cases, the designated flight item is either a new build, or F/G spare hardware. Spares for the Out-of-Ecliptic spacecraft would come from the prototype inventory. The new build/procurement items include the structure, RTGs and RHUs, propulsion subsystem, thermal subsystem, harnesses, and selected electronic units.

4.6.3 Refurbishment Cost Effectiveness

The plan for new hardware replacement vs. rework of prototype or spare hardware is conservative as presented. Reworked prototype hardware are listed as flight items mainly in cases where redundancies exist. The cost savings of this approach over a totally new spacecraft derives mainly from the use of F/G spare electronic units for flight inventory, and prototype hardware for spare inventory. Further savings could be realized by taking advantage of the payload weight margin available with the Titan III D configuration, and selectively adding additional structure at critical locations. This would allow unqualified use of the prototype structure and eliminate the cost of new builds for this subsystem. The potential savings for full refurbishment vs. new hardware are estimated to be approximately 40% of the cost to produce new hardware from an existing design. The refurbishment plan for the prototype spacecraft presented here calls for a hardware mix of approximately 30% new, 60% spare, and 10% prototype. Calculated on the basis of

TABLE IV-8.

SPACECRAFT REFURBISHMENT PLAN - PIONEER H

Subsystem/Item	Source of Flight Unit	Source of Flight Spare	Comments
Structure			
Platform	New Build		Possible cost savings by selective "beefing" of prototype structure
Antenna Starts	New Build		
RTG Booms	New Build		
Magnetometer Boom	New Build		
Antennas/Feeds			
High Gain Dish	Prototype		Range test required
High Gain Feed	New Build		
Feed Mechanism	New Unit	F/G Spare	
Med Gain Ant.	New Unit	Prototype	Range Test required
Low Gain Ant.	New Unit	F/G Spare	
Thermal Subsystem			
Louver Assy.	New Unit	Prototype	
Insulation and Coatings	New Install.		
RHU's	New Units	New Units	
Power			
RTG's	New Units		
Inverter #1	F/G Spare		
Inverter #2	Prototype	Prototype	
PCU	F/G Spare	Prototype	
CTRF	F/G Spare	Prototype	
Battery	New Unit	New Unit	
Communications			
Transmitter Driver #1	F/G Spare	Prototype	
Transmitter Driver #2	Prototype	Prototype	
Receiver #1	F/G Spare	Prototype	
Receiver #2	Prototype		
Conscan	F/G Spare	Prototype	
TWTA #1	F/G Spare	Prototype	
TWTA #2	Prototype	Prototype	
CDU	F/G Spare	Prototype	

TABLE IV-8. - continued -

Subsystem/Item	Source of Flight Unit	Source of Flight Spare	Comments
Diplexer #1	F/G Spare	Prototype	
Diplexer #2	Prototype		
RF Switch #1	New Unit	F/G Spare	
RF Switch #2	New Unit	F/G Spare	
RF Cables	New Build		
Data Handling			
DTU	F/G Spare	Prototype	
DSU	F/G Spare	Prototype	
DDU #1	F/G Spare	Prototype	
DDU #2	Prototype		
Attitude Control			
CEA	F/G Spare	Prototype	
SSA	F/G Spare	Prototype	
DSA	F/G Spare	Prototype	
SRA	New Unit	F/G Spare	
Propulsion			
Propellant Tank	New Unit		
Valves	New Unit	F/G Spare	
Thruster Assy's	New Unit	F/G Spare	
Catalyst Bed	New Unit	F/G Spare	
Plumbing	New Install.		
Other			
Harness	New Build		Connector wear
Ordnance	New Items	F/G Spare	

these percentages, the refurbishment plan as shown in Table IV-8 represents a potential cost savings of between 20 and 30% below that of a completely new system. This does not include the savings due to use of prototype hardware as spares or potential salvage of the present prototype structure.

4.6.4 Prototype Availability

The spacecraft systems contract for the F/G program with TRW calls for the prototype spacecraft to be maintained for flight backup and special test purposes through the G launch. It has been stated however, that the engineering model spacecraft would adequately serve the requirement for a test system during this period. Use of the prototype as a backup for either F or G is not seriously contemplated. Assuming a favorable program decision on these two points, the prototype spacecraft would complete its function for the F/G program at the end of the system qualification tests. This date is now estimated to be in August 1971.

4.7 DESIGN ALTERNATES

Consideration of the Out-of-Ecliptic mission requirements and the present Pioneer capabilities has led to an examination of potential design changes which offer either an improvement to specific capability or to overall mission reliability. These potential changes are discussed in this section.

4.7.1 Low Power TWT

The 8-watt TWTA is a major spacecraft electrical load, comprising approximately 40% of the total spacecraft requirement. Therefore, it is a potential candidate for reducing the power drain. The impact of the reduction in RF radiated power, moreover, is offset by the decreasing Earth-spacecraft communication range occurring during the latter portion of the mission. At distances less than 3 A.U., a 4-watt TWT would still provide the maximum data rate capability of 2048 BPS.

The technical modifications are predicated on changes to the existing 8-watt TWTA design by the Watkins-Johnson Company for the following reasons:

a. A space-qualified off-the-shelf TWTA, capable of meeting frequency, magnetic, weight, efficiency, and interface requirements does not exist.
b. Development of a TWTA to meet the spacecraft requirements by a different vendor is impractical because the changes to the W-J design, which is proprietary, are minor.

The overall DC to RF efficiency which may be expected from the modified design is approximately 24% versus 26.5% for the 8-watt TWTA. Therefore, approximately 16.5 watts of primary power at 28 VDC is required to support the 4-watt transmission capability.

A mandatory change in the tube structure is an increase in the cathode-anode spacing. This is essential to ensure the operating anode potential is more positive than the helix potential (by approximately 30-50 volts). Failure to achieve this adversely affects the tube life by causing an ion stream to impinge on the cathode and deplete the coating.

A second highly desirable change involves geometrically scaling down the cathode structure in order to reduce filament current and thereby optimize efficiency. A significant improvement in tube efficiency, 32% versus 28.5%, is predicted for the modified cathode. This is a relatively minor change using the same basic parts as the 8-watt design and requiring no mechanical changes.

There are essentially no modifications required in the converter portion of the TWTA. The specific tube element requirements can be met with appropriate windings on the existing high voltage transformer.

This is identical to the procedure presently followed in matching the requirements of each 8-watt tube to a specific converter. The converter efficiency is degraded slightly, from approximately 91% to 89%, due to a higher percentage of core losses with the reduced load consumption.

Minor changes to the telemetry signal conditioning circuitry may be desirable to accommodate possible range changes in the telemetered parameters. If required, these modifications are minor, generally consisting of component substitutions.

4.7.2 Redundant Sun Sensor in 0-12 deg. Sector

The Sun sensor is divided into 3 sectors where all but the low angle (0-12°) portion is redundant. For the F and G missions, redundancy in spacecraft attitudes with the Sun line close to the spin axis, is provided by the SRA (Stellar Reference Assembly). During portions of certain Out-of-Ecliptic trajectories, the principal star Canopus will be outside the viewing cone of the SRA, thus placing full dependence on the single Sun sensor channel. The redundant element would fit within the present SSA package. Additional weight is estimated at 0.3 lbs. and additional power would be approximately 0.25 watts.

4.7.3 RTG Modifications

Increased fin size for the RTGs would lower the hot junction temperature and reduce the rate of degradation of the thermocouples, according to the RTG contractor, Isotopes, Inc. The spacecraft configuration and structure has been examined to ascertain what fin area increase and weight increase could be tolerated without critically affecting present design. An analysis of structural limitations shows that an RTG weight increase of 10% can be reasonably accommodated. Surrounding structure limits total fin span to approximately 20 inches (vs. the present 16 in). The resulting increased heat rejection capacity would serve to lower the thermo-electric hot junction temperature by an estimated 20°F. Isotopes predicts this would reduce the rate of degradation by approximately 0.05 watt/month unit.

4.7.4 Battery

Investigation of the Pioneer F/G silver cadmium battery design has provided information which suggests that performance is not reliably predictable beyond a 2.5 year lifetime. Alternatives have been posed which include:

1. Replacement of the Ag Cd battery with a Ni Cd of comparable performance characteristics.
2. Fly the present Ag Cd battery, but provide more controlled thermal conditions for an extended lifetime.
3. Fly the Ag Cd battery, but provide for an in-flight activation to be performed later in the mission when RTG power has degraded and the battery is required for peak loads.

Ni Cd batteries have operating life expectancies which are compatible with the Out-of-Ecliptic requirements. The most significant problem encountered with the Ni Cd battery is the magnetic property of Ni Cd cells. A battery otherwise suitable would produce a field of estimated 0.1 gamma at the magnetometer, when the spacecraft is magnetized (The specification for the field at the magnetometer allowed for the entire spacecraft is 0.25 gamma). Shielding techniques and cell orientation arrangements with the battery hold some promise for a further reduction of its contribution to the total field, however, further investigation would be required to be certain of results. In addition to the magnetics problem, a comparably performing Ni Cd battery would likely represent a weight increase of 2-3 lbs, and require a new charge control system.

Investigation was also made of possible improvement to the present Ag Cd battery lifetime through use of more sophisticapted in-flight temperature control. Battery sizing and predicted capacity decay for the present battery is based primarily on Goddard and Convair battery flight data. One of the chief determinates has been the Goddard IMP E flight of 2-1/2 years at an average temperature of 10°C with an eventual 80% loss in capacity. A capacity decay of 50% is predicted for Pioneer F/G by minimizing pre-flight time and going to a lower average temperature (0° to -5°C). There is no doubt that reduced temperature will reduce the rate of chemical decay or capacity loss, but the F and G prediction has not yet been proved. Furthermore, development tests have shown that the present battery becomes marginal with respect to the present charge control system at 0°F, and that the apparent cell capacity drops significantly at reduced temperature (from 5 to 2 AH for a new cell in going from 70° to 0°F). From this, it is preliminarily concluded that the present battery could be made to last 4 years by dropping the temperature to -10°C or lower, but in dropping to that temperature the lower limit of operation for this size cell is approached. Therefore, a new temperature control technique, along with charge control changes and/or cell size increase may be necessary to reach 4 years of life. (Increased cell size would result in improved capacity at the low temperature and at the same time permit charging rates approaching the present battery rates.)

A very preliminary examination indicates that a thermostatic control system would probably best suit estimated need; this system would permit temperature control at ±5°F, would avoid the decaying temperature profile of the present passive system (and its inherent hotter temperatures in the early and late phases of the mission), and avoid excessively cold temperatures at Jupiter. A small, (approximately 1 watt) heater would be necessary on a discontinuous basis, with the battery being well insulated to reduce heat exchange with the spacecraft. Development testing to optimize cell size and/or charge rates along with temperature would be needed.

It should be pointed out previous Ag Cd life/cycle testing at low temperatures by NASA produced several early failures. It is believed that these failures were due to the charge control techniques used, but there may be complications in low temperature Ag Cd operation for extended periods.

The possibility of activating an Ag Cd battery at some time well into the mission was investigated. The Eagle-Pitcher Company who furnish the Pioneer F/G battery was contacted. They have built Ag Zn batteries of this type, but these are a one-time, one-shot battery with a large output for a short duration, and are therefore, not satisfactory for this application. Eagle-Pitcher has not built a remotely activated Ag Cd battery but believe it is feasible. Concern was expressed, however, with regard to the purging or removing of the electrolyte in the manifold during activation and a development program would most likely be required.

4.7.5 Radiation Shielding

Shielding design has not been addressed in this study, largely because it is possible to target further away from the deep Jovian radiation belts and still achieve maximum inclinations. However, this does impose a penalty of longer total trip time. It may be more desirable to utilize some of the payload weight margin (available with the Titan Centaur) in shielding materials for selected components. The spacecraft power inverters, with their relatively low damage thresholds, would be possible candidates.

5.0 EXPERIMENTS INSTRUMENTS

The basic instrument complement considered for the Pioneer Out-of-Ecliptic Mission is the present F/G payload. As noted in Section 2.0, these instruments provide a strong interplanetary particles and fields capability well suited to the Out-of-Ecliptic exploration. Further, some of the primarily planetological instruments will have interplanetary importance, such as zodiacal light measurements by the Imaging Photopolarimeter.

The requirements of the F/G instruments are described and the availability of flight units for an Out-of-Ecliptic mission is discussed.

Selected alternatives to the F/G payload have been analyzed in terms of spacecraft interface impact and schedule feasibility. The types of payload changes which appear practical in the time period for a 1974 launch are presented.

There has been no attempt to specify an optimum set of Out-of-Ecliptic experiments, although it is readily seen that the addition of instruments for measurement of cosmic dust, electric fields, and radio noise would provide important enhancement to the present F/G complement. Rather, the study attempts to show the capability of the F/G system to accommodate classes of payload changes, which might include additions or replacements involving the experiments cited above.

5.1 PIONEER F/G EXPERIMENT INSTRUMENTATION

The scientific payload for the Out-of-Ecliptic mission has been assumed to be the present Pioneer F/G instruments. The particular investigations represented by these eleven instruments include:

a. Magnetic Fields
b. Plasma
c. Changed Particle Composition
d. Jovian Charged Particles
e. Cosmic Ray Energy Spectra
f. Ultraviolet Photometry
g. Imaging Photopolarimetry
h. Jovian Infrared Thermal Structure
i. Asteroid/Meteoroid Astronomy
j. Meteoroid Detection

5.2 INSTRUMENT WEIGHTS

The F/G Instruments weigh a total of 63.5 lbs. Individual instrument weights are as follows:

Instrument	Weight (lbs)
JPL/Helium Vector Magnetometer	6.0
ARC/Plasma Analyzer	12.1
Chicago/Charged Particle Instrument	7.2
U. Iowa/Geiger Tube Telescope	3.6
GSFC/Cosmic Ray Telescope	7.0
UCSD/Trapped Radiation Detector	3.8
USC/UV Photometer	1.5
Arizona/Imaging Photopolarimeter	9.4
CIT/Infrared Radiometer	4.4
G.E./Asteroid-Meteoroid Detector	7.0
LaRC Meteoroid Detector	3.5

5.3 INSTRUMENT POWER REQUIREMENTS

Individual instrument power requirements are as follows:

Instrument	Nominal Power (watts)	Peak
JPL/Helium Vector Magnetometer	4.5	4.5
ARC/Plasma Analyzer	4.5	4.5
UC/Charged Particle Instrument	2.1	2.1
UI/Geiger Tube Telescope	0.7	0.7
GSFC/Cosmic Ray Telescope	2.1	2.4
UCSD/Trapped Radiation Detector	2.9	2.9
USC/Ultra Violet Photometer	0.6	0.6
UA/Imaging Photopolarimeter	2.8	3.5
CIT/Infrared Radiometer	1.3	1.8
GE/Asteroid-Meteoroid Detector	2.1	2.8
LaRC Meteoroid Detector	0.6	1.0
	24.2	26.8

5.4 EXPERIMENT VIEWING REQUIREMENTS

a. Instrument pointing with respect to the sun

b. Instrument pointing during Jupiter encounter

5.4.1 Solar Viewing

The F/G viewing instruments are designed with the intention that the Sun is normally not within the field. An exception is the Plasma Analyzer, which with its present mounting, has requirements that define the direction of the Sun with respect to the spacecraft spin axis (ϕ) as follows:

$10° > \phi < 30°$ Good

$5° > \phi < 40°$ Acceptable

Plots of the Earth-Spacecraft-Sun angle, which represents ϕ, are given in Section 3, Figures 3-16, 3-20 and 3-30. From these plots, ϕ is shown to be within the prescribed bounds.

5.4.2 Planetary Viewing

Figures 5-1 and 5-2 show the Earth-Spacecraft-Jupiter angles for planetary encounter for both prime and alternate targeting. The UV Instrument as presently mounted, will not see the planet. The IPP and IRR instruments will have reasonable viewing opportunities as shown.

5.5 INSTRUMENT AVAILABILITY

A survey of current F/G instrument status and planning was made to determine availability of hardware for an Out-of-Ecliptic mission. A flight-worthy version of each of the 11 Pioneer F/G scientific instruments will be produced as backup for the G flight. Four of these units will have been integrated on the prototype spacecraft, subjected to the prototype spacecraft qualification test program, then refurbished and acceptance tested to flight hardware standards. The Design Verification Unit (DVU) for each instrument could be refurbished and provided as the flight spare.

On the basis of this approach, no new instruments would be required for an Out-of-Ecliptic Mission.

5.6 ALTERNATIVE PAYLOADS

Investigations were made of alternative payloads which are compatible with a 1974 launch schedule. These alternatives are necessarily constrained to a very limited impact on present spacecraft design. The alternatives are classified as follows, more or less in order of increasing program impact:

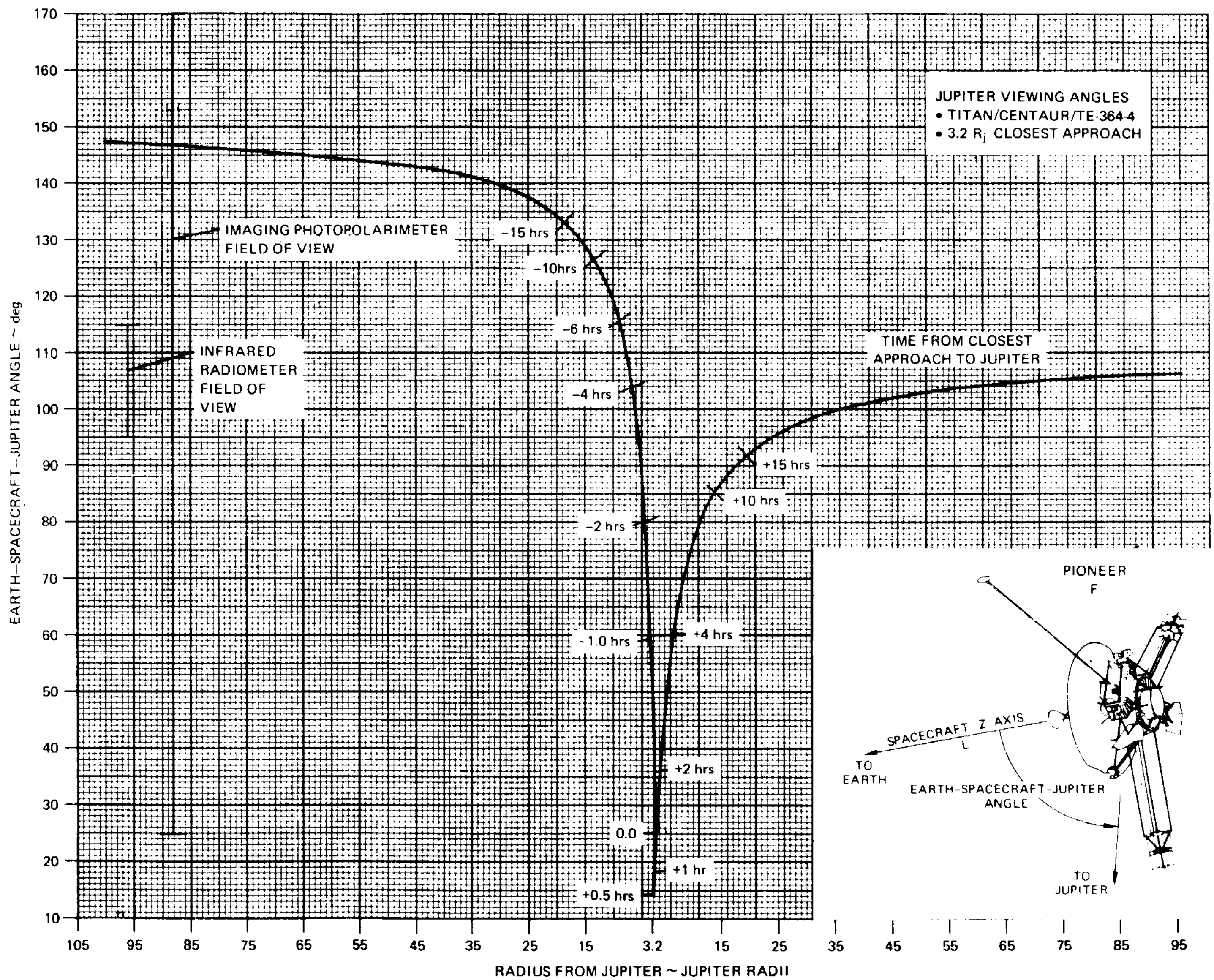

FIGURE 5-1. – EARTH-SPACECRAFT-JUPITER ANGLE

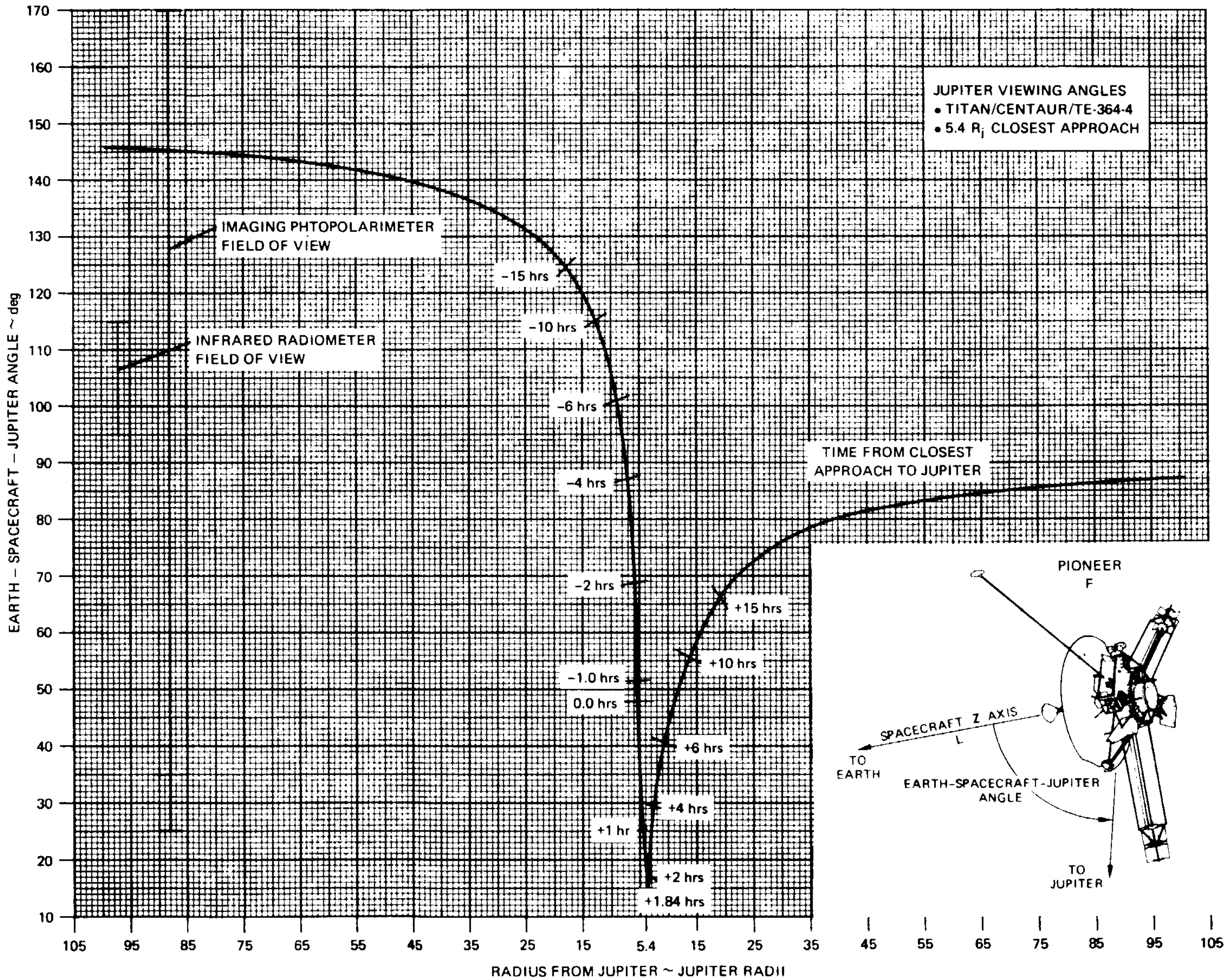

FIGURE 5-2. – EARTH-SPACECRAFT-JUPITER ANGLE

a. Modify present instrument mounting for improved planetary viewing.
b. Perform limited internal modifications to present instruments.
c. Replace present instruments with new instruments.
d. Add new instruments to the present complement.

Considerations for these changes with regard to schedule feasibility and the spacecraft interface are assessed in the subsequent paragraphs:

5.6.1 Instrument Mounting Changes

This change classification would apply to the UV experiment, where a 30-40° reorientation of the instrument would be required to obtain a view of the planet during swing-by. This would involve a simple mechanical mounting change, which should not affect other areas of the spacecraft interface.

5.6.2 Instrument Internal Modifications

It is intended that this category be limited to measurement range adjustments, detector changes, etc., within the bounds of present packaging, with minor increases to power and weight permitted. This is a difficult type of change to control, and would require careful attention to insure that the instruments were delivered on schedule and without introducing interface problems, such as EMI, in the process.

5.6.3 Replace Present Instruments with New Instruments

This category of change becomes very sensitive to the particular instruments involved. What is thought to be possible with 1974 launch schedule constraints are replacements where the instruments involved share common interface requirements, including viewing, weight, power and signal characteristics. A very important consideration would be that the replacement instrument be already developed and available so that integration requirements could be established with some certainty at the initiation of spacecraft refurbishment. The constraint of no subsystem level changes in the spacecraft is emphasized.

5.6.4 Add New Instruments

This category represents potentially the highest risk and largest impact on the spacecraft of the payload changes considered. As noted in section 4.4, the spacecraft has the capability to carry and support additional payload and the Titan Centaur affords considerable payload margin. The ultimate constraint on new additions is schedule and cost.

From the spacecraft schedule standpoint, this means no changes at the subsystem level. The instruments would be required to fit within present capability as described in section 4.4. As stated above in 5.6.3, the new instrument should either be in existence now or simple enough to technically justify waiving the normal instrument development program. Mounting positions for new instruments are somewhat limited, however, there are locations external to the equipment compartment which afford good side-looking fields of view. Instruments requiring a direct view of the Sun pose a more serious problem, because of the positioning of the high-gain antenna. Viewing ports in the dish are now installed for some instruments, and additional holes could be provided but with some degradation to antenna performance. Changes in this category would need careful planning and control to avoid excessive risk to the spacecraft schedule.

6.0 PROGRAM IMPLEMENTATION

The Pioneer H Out-of-Ecliptic Mission is a logical and compatible follow-on to the Pioneer Program. Use of the F/G Prototype Spacecraft and F/G Science Instruments permit a reasonable schedule for the 1974 launch opportunity with a FY 1973 start. The Pioneer H Mission benefits from the development and fabrication of both spacecraft, experiment instruments and GSE, already accomplished by the F/G project. Costs for the H Mission are consequently lower than would otherwise be experienced.

Since Pioneer H technology and operations will be similar to Pioneer F/G, it will be efficient to phase personnel from the F/G Missions to the H Mission. Furthermore, since the same technology and experience are required for the missions, the combined effort can be conducted with fewer total personnel since subsystem people can serve all missions simultaneously. As a result of these factors, Ames can manage the H Mission without any increase in personnel.

Figure 6-1 shows the Program Schedule.

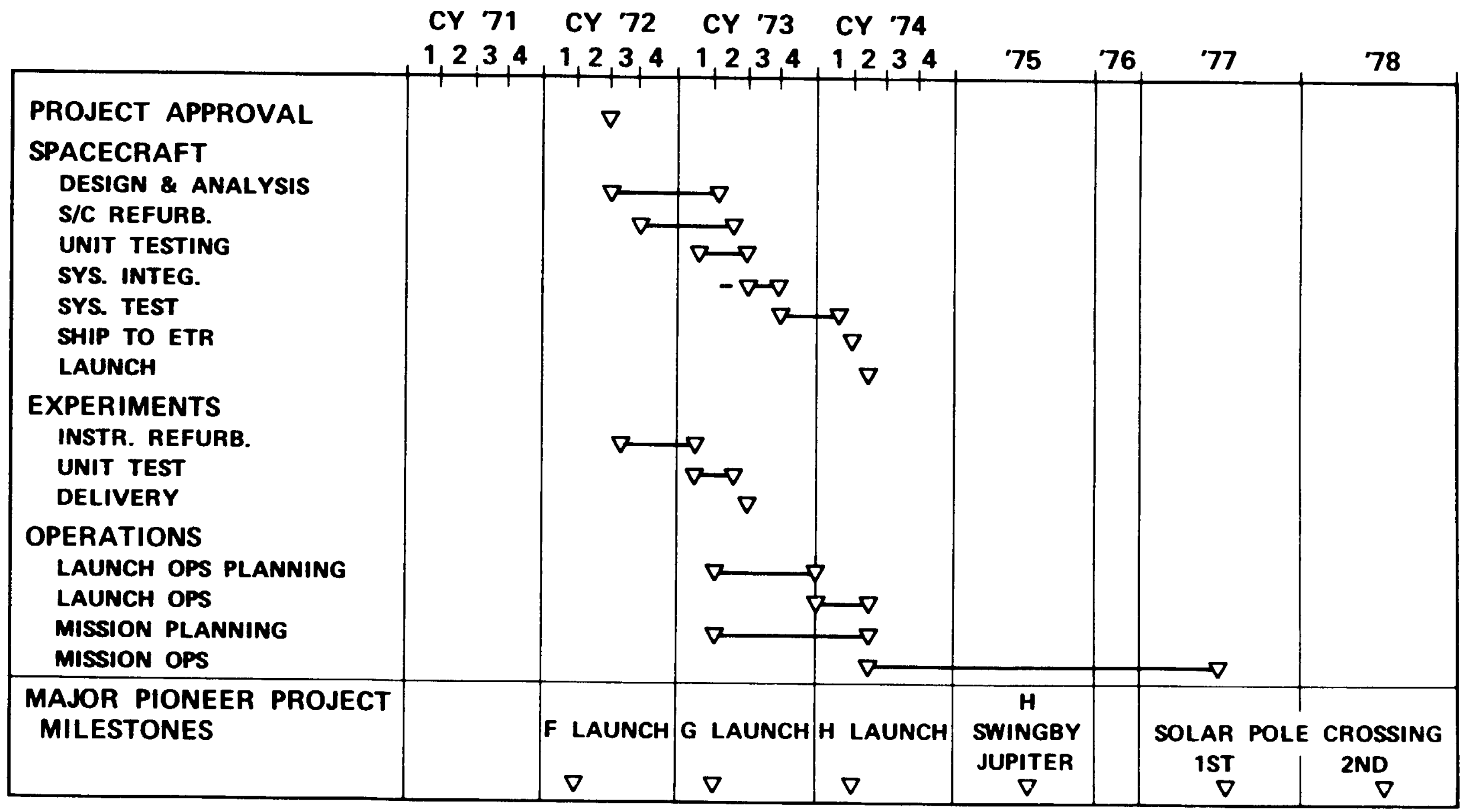

Figure 6-1.- Pioneer H - Schedule for 1974 Launch